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AIRCRAFT POWER PLANTS

A Report of the AAF Scientific Advisory Group

by

FRANK L. WATTENDORF
H. S. TSIEN   POL DUWEZ

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Aircraft
POWER PLANTS

A REPORT PREPARED FOR THE AAF SCIENTIFIC ADVISORY GROUP

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The AAF Scientific Advisory Group was activated late in 1944 by General of the Army H. H. Arnold. He secured the services of Dr. Theodore von Karman, renowned scientist and consultant in aeronautics, who agreed to organize and direct the group.

Dr. von Karman gathered about him a group of American scientists from every field of research having a bearing on air power. These men then analyzed important developments in the basic sciences, both here and abroad, and attempted to evaluate the effects of their application to air power.

This volume is one of a group of reports made to the Army Air Forces by the Scientific Advisory Group.

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PART I

GAS TURBINE PROPULSION

By

FRANK L. WATTENDROF

PART I

GAS TURBINE PROPULSION

DECEMBER 1945

INTRODUCTION

Gas-turbine propulsion is used as a general term covering the various means of propulsion incorporating a gas turbine as its primary component. Included under gas-turbine propulsion are the following systems:

(1) Turboprop, or gas turbine driving a propeller.

(2) Turbofan, or gas turbine driving a ducted fan.

(3) Turbojet, or gas turbine plus jet reaction.

In the present report brief comments will first be made on the reciprocating engine, compound engine, free piston, and various gas-turbine systems. Then the scientific background of some of the propulsion systems will be discussed from the point of view of stressing future possibilities of development.

The primary purpose of this report is to show ways in which the gas turbine is basically capable of considerable improvements toward increasing its potentialities for high-speed aircraft application. In order to obtain improvements of this nature, greater emphasis on basic and applied research is needed. This report stresses in particular the items on which research and development should pay the greatest dividends.

RECIPROCATING ENGINE

The chief advantage of the reciprocating engine at the present time is in low specific fuel consumption, especially at partial load. This makes it particularly applicable to long-range flight at present cruising speeds. Typical reciprocating engines now in use have specific fuel consumptions which are in the neighborhood of 0.7 lb/bhp-hr at full-rated rpm, decreasing to about 0.45 lb/bhp-hr at cruising conditions.

The principle characteristics of several current and proposed reciprocating engines of higher horsepower of interest for comparison with gas turbines, are given in Table I.

The largest reciprocating engine now under development is the 5000-hp Lycoming R-7755. One of the chief aims of this engine is fuel economy for long-range bomb-

per application. The estimated minimum specific fuel consumption is approximately 0.37 lb/bhp-hr, the improvement being associated largely with increased compression ratio. However, it is noted that there is no improvement in either specific weight or power per unit frontal area of the R-7755 over the R-4360. The ratio of dry weight to take-off power is 1.21 lb/hp for the R-7755 as compared with 1.14 lb/hp for the R-4360 and the ratio of take-off power to frontal area is 178 for the R-7755 as compared with 198 for the R-4360.

Further increase of reciprocating-engine power involves further increase in cylinder volume, which is already at a critical stage as regards number and arrangement of cylinders, complexity of the drive, excessive frontal area, and difficulty of cooling. There are development projects at present aimed at increasing the output of the R-3350 to 3200 hp and the R-4360 to 4300 hp.

Some increase in fuel economy can reasonably be expected by utilizing higher pressure ratios, improved fuels, improved metallurgy, etc., but engine manufacturers consider that it would be difficult to attain a specific fuel consumption essentially lower than 0.35 lb/bhp-hr.

A serious limitation of the reciprocating engine for high-speed flight is the cooling problem, as pointed out in the discussion on airplane designs by Sears, Ashkenas and Hasert (Part II of *Aerodynamics and Aircraft Design,* another report of the AAF Scientific Advisory Group). The adiabatic rise in temperature due to impingement of the air on the engine surfaces is proportional to the square of the air velocity, and would reach a value of 90°F at 700 mph, and 180°F at 1000 mph. This cooling difficulty coupled with the relatively large frontal area, would act as a serious drawback to the application of the reciprocating engine to supersonic flight.

The motorjet or Campini system, whereby a reciprocating engine drives a ducted fan, has several advantages, especially that afterburning can be used for temporary increase in thrust, but the basic limitations are those associated with the reciprocating engine.

COMPOUND ENGINES

The compound engine represents the beginning of a transition between the conventional engine and the gas turbine. Compounding is an attempt to recover a considerable portion of exhaust energy by operating some form of a gas turbine which feeds back into the main power drive. Several projects now under consideration are:

1. Wright Aeronautical Corporation proposes a system in which the exhaust blows forward through turbine buckets located on the tips of cooling fan blades in the front nose cowl ring. The turbine drives the cooling fan, and the surplus power is transmitted to the propeller shaft through a gear transmission. By this means, it is estimated that the fuel consumption can be reduced from 10 to 20 percent.

2. Allison V-1710-127. An exhaust turbine is mounted on the rear of the auxiliary stage supercharger, and is connected to the crankshaft system by means of an extension shaft.

3. The General Electric Company at West Lynn have a development project for an exhaust gas turbine feeding back into the crankshaft of a R-4360 engine.

FREE-PISTON ENGINE

In the compound engine described in the preceding paragraphs, the pistons of a reciprocating engine are used partly to drive a crankshaft and partly to serve as a gas generator to drive a gas turbine. The so-called free-piston engine represents the extreme in the compounding principle, in that the pistons are used solely as gas-generators, and the products of combustion are used entirely to run a gas turbine.

The main principle of the free-piston engine is the utilization of floating pistons which, instead of driving connecting rods and crankshafts as in conventional engines, generate compressed and heated gases by recoiling to and fro between cushions of charged air, with combustion taking place at the time of maximum compression of each charge. The products of combustion are then available either to pass directly through a discharge nozzle, furnishing jet propulsion, or they may pass through a turbine, and generate power for driving a propeller or ducted fan. For the gas-turbine application, one of the chief advantages claimed for the free-piston engine is low specific fuel consumption, which is associated with the fact that combustion takes place at a relatively high value of instantaneous pressure. Since partial expansions take place in the cylinder after combustion, the temperature of the gases entering the turbine is not excessively high. The free piston can also be used to compress air which is mixed with hot gases, thereby resulting in a cooler mixture. These factors are favorable for turbine wheel construction and life. In addition, the partial load characteristics are more favorable than the normal gas turbine.

The successful applications of the free-piston engine so far have been to stationary power plants or marine engines, where weight and bulk were not important factors. Professor Junkers had been experimenting in Germany with the free-piston principle for several decades, and the Junkers Company has been making a free-piston marine engine. Pescara in France has developed stationary power plants of the free-piston type. Sulzer Brothers in Switzerland also build free-piston power plants.

The Japanese Navy, in 1940, became interested in applying the free-piston engine to gas turbine - propeller combinations for aircraft propulsion, and in 1941 a model free-piston gas turbine was placed in operation. It soon became apparent, however, that too much development work would be needed before it could be useful in the war effort, and it was later abandoned in favor of copying German turbojets.

In Germany, O. Lutz of the Luftfahrtforschungsanstalt Hermann Göring was attempting to develop a free-piston engine for aircraft application. This engine was claimed to be essentially lighter in weight and smaller in bulk than the Pescara and Junkers types. This was accomplished chiefly by having a series of pistons rotate and oscillate within a torus-shaped housing. In a working prototype there are two banks of three pistons each, which rotate about a central axis. Although both groups of pistons rotate at the same mean speed, they have a superimposed oscillatory motion of such nature that one piston group is always oppositely phased to the other. In this way there are six working spaces, of which three are expanding while the other three are contracting at the same rate. Valves are simple slots which require no further mechanism. By the end of the war, the prototype had not been tested enough to indicate its feasibility.

With regard to possible aircraft application of the free piston, it is believed that a specific fuel consumption of 0.35 lb/bhp-hr may be attainable. Considerable re-

search on the subject is needed to find the best piston arrangement and motion synchronization to give a high cycle efficiency and power output without excessive size and weight.

GAS TURBINE

1. *Description of the Turboprop.*

A simple turboprop or gas turbine-propeller combination is shown schematically in Fig. 1. Air enters the inlet duct under local atmospheric conditions, and is compressed by a compressor of either the axial or centrifugal type. The original pressure and temperature of the entering air are denoted by P_1 and T_1 respectively, and the values change through the inlet duct to P_2 and T_2 at the upstream face of the compressor. Through the compressor, the pressure is increased to P_3 and the temperature to T_3 respectively. Combustion takes place in the combustion chamber into which fuel is supplied through nozzles. Through combustion, the pressure drops somewhat to P_4, while the temperature is increased by the combustion process to T_4. The products of combustion, at elevated pressure and temperature, pass through the turbine, through which the pressure drops to a value P_5 and the temperature to T_5. The exhaust gases then pass through the tail pipe where they are finally discharged to the atmosphere under conditions P_6 and T_6. The compressor and turbine are usually interconnected by the same shaft, and in a normal turboprop part of the turbine work is used in driving the compressor, while the remainder goes into the propeller drive. Some residual energy goes into the jet, which is also utilized for propulsion. In the pure turbojet, which will be discussed in a later section, the turbine takes just enough energy from the gas to drive the compressor, while the rest of the energy goes entirely into the jet and is the sole means of propulsion.

2. *Thermodynamic Cycle.*

The simple gas-turbine cycle is represented on the pressure-volume diagram in Fig. 2a. A - B represents the adiabatic compression in the compressor, B - C is combustion at essentially constant pressure, and C - D is adiabatic expansion in the turbine. The net work performed by the cycle is given by the area under the curve. This same cycle in the temperature-entropy or the enthalpy-entropy diagram is shown in Fig. 2b. The full lines show the theoretical cycle with adiabatic expansion, and the dotted lines represent the actual cycle where the deviation from the adiabatic is due to losses in the compressor and turbine

From such diagrams, the specific net power and the specific fuel consumption can be estimated for various assumed conditions of pressure ratio, turbine-inlet temperature, compressor efficiency, turbine efficiency, etc. This has been done by M. Alperin for a systematic series of values of the different factors, for the purpose of studying various influences on gas-turbine performance, as discussed in the following paragraphs.

3. *Influence of Turbine-Inlet Temperature.*

For the development of future improved gas turbines, it is of importance ot study the influences of various factors on the theoretical performance of the gas turbine. One

significant factor is the turbine inlet temperature T_4. It is seen from the cycle diagram of Fig. 2 that higher temperatures result in greater work output. However, as far as a theorectical cycle without losses is concerned, the fuel consumption would increase proportionately with the net output so that the specific fuel consumption would be independent of turbine-inlet temperature. However, for the actual cycle with losses, increased turbine-inlet temperature has generally a beneficial effect on fuel consumption.

This is shown in Fig. 3, for temperatures of 1600°F, which have already been exceeded; 2000°F, which several current units almost attain, and 2600°F for a future goal. A flight Mach No. of 0.8 is used for this example. The assumed compressor and turbine efficiencies of 85% have already been attained in axial flow units.

It is seen that the minimum specific fuel consumption shows a moderate decrease with increasing temperature. The influence of turbine-inlet temperature on the power output is much greater, as shown in Fig. 4, for the same range of temperatures. It is seen that the power output would be increased 60% at (compr. ratio = 7) 2000°F and over 180% at (compr. ratio = 10) 2600°F for the same mass flow, providing the pressure ratio were also increased for the higher temperatures. The magnitude of the power increase emphasizes the need for intensive support of high-temperature materials development. — but also need for higher compression ratios to utilize effects of increased temp.

4. *Influence of Compressor and Turbine Efficiency.*

The influence of efficiency on specific fuel consumption as a function of flight speed is shown in Fig. 5 for a constant pressure ratio of 6:1. The corresponding curves for shaft horsepower are shown in Fig. 6. The values of 85% for component efficiencies are fairly representative of current axial flow units. Laboratory tests of experimental units have indicated component efficiencies of 90% to be within reach. As an ultimate goal, the values of 94% for compressor efficiency ,and 96% for turbine efficiency, were used because these values assume that all secondary losses have been eliminated, and only the unavoidable skin friction remains; in other words, they represent ceiling values. As a matter of interest, the ideal curves for 100% efficiency are also included.

This gain in fuel economy and shaft horsepower emphasizes the need for components development, involving new blade shapes, study of mutual interference between blades, three-dimensional flow, compressibility effects, and the influence of centrifugal force on boundary layer flow. All these subjects relate to the important field of high-speed flow in rotating machinery, which should be given much more attention in the future.

5. *Intercooling.*

One means of improving the power output of a gas turbine is by intercooling between compressor stages. By this method the power required by the compressor to obtain a given pressure ratio is less, which leaves more of the turbine output for the propeller. The nature of intercooling is such that its effectiveness increases with pressure ratio.

The pressure ratios of most current units are too low to warrant incorporation of intercooling, and its utility is greater for gas turbines with high pressure ratios. It is likely to become of increasing importance since pressure ratios are becoming pro-

gressively higher. Interviews with manufacturers indicate that the majority did not think that intercooling paid for the increased weight and complication, except for units of high pressure ratio.

6. Regeneration.

One of the most effective means of improving cycle efficiency and decreasing specific fuel consumption is by the use of regeneration. The reasoning behind this is that the exhaust gas leaving the turbine still has an essential temperature head. This heat may be partially recoveredby the incorporation of a heat exchanger in the turbine exhaust. If this heat is used to preheat the air entering the combustion chamber, it is obvious that less fuel will be eneded to raise the temperature of the combustion gases to the allowable limiting value at the turbine inlet. By this means, it is estimated that a saving of between 20 and 30 percent in the specific fuel consumption is possible. Against this advantage must be weighed the disadvantage of increased weight, pressure loss, and complications. Wright Aeronautical Corporation engineers, among others, have made a theoretical study of regeneration, including weight estimates.

Sample estimates for a typical 5000-hp gas turbine are:

Without Regeneration	
Dry Weight	4000 lb
Specific Fuel Consumption at Sea Level	.80 lb/hp/hr
Specific Fuel Consumption at 20,000 ft	.58 lb/hp/hr
With Regeneration	
Dry Weight	5600 lb
Specific Fuel Consumption at Sea Level	.56 lb/hp/hr
Special Fuel Consumption at 20,000 ft	.41 lb/hp/hr

One of the chief factors hindering development of the regenerative system is that this type of heat exchanger for aircraft application is in the early stages of development. A German project for developing a ceramic heat exchanger was being conducted at Göttingen, and should be reviewed for possible application. General research on the development of light and efficient heat exchangers with low pressure loss for aircraft application is recommended.

7. Reheat.

A further increase in performance of the gas-turbine cycle can be obtained by dividing the turbine into several separate stages and reheating the gas by combustion between the stages. Both power and fuel economy are improved. Against this gain the increased complication and weight must be weighed. Reheat for aircraft gas turbines is relatively unexplored, because lightness and simplicity were necessarily prerequisite to the first aircraft applications. All such methods of improvement should be explored for the future.

8. Separate Propeller Turbine.

The simple gas turbine has the fundamental characteristic of having its performance improve with rotational speed. This is opposite in nature to the reciprocating engine, which fact makes the normal gas turbine relatively more disadvantageous

under cruising conditions than at maximum output. One method of improving gas turbine performance at partial load is obtained by using two turbines, one driving the compressor only without giving any net power, and the other supplying power solely to the propeller. The improvement at partial load is brought about by the fact that the turbine driving the compressor can always be operated at the most suitable speed for the compressor, independently of the speed of the turbine driving the propeller. It is obvious that the two-turbine system is more complicated than the single-turbine system; however, it might be justified for large, long-range airplanes which operate a major portion of the time at cruising speed. For such application, the advantage of the two-turbine system must be compared with alternative plans such as utilizing a smaller single gas turbine which has its maximum speed at cruising conditions, and obtaining higher speeds for short periods by supplemental burning or other temporary methods of increased performance, or in a multi-engine aircraft by having several units which are closed off under cruising conditions and only used for high speeds.

9. *Closed Cycle.*

In the closed cycle as developed by J. Ackeret and C. Keller for Escher Wyss in Zürich, the gas passing through the compressor and turbine is recirculated in a closed circuit, while combustion takes place externally and transmits heat to the gas through a heat exchanger. This cycle offers several advantages from the theoretical point of view; for instance, since the cycle is enclosed, it may be made gas tight and operated at any desired pressure level. By using air under pressure, it is obvious that a higher output of power may be obtained with smaller dimensions of compressor and turbine. Also, there is much better control of the temperature distribution entering the turbine, which enables the turbine to operate at a somewhat higher level of mean temperature than would otherwise be the case. A schematic diagram of a closed cycle is shown in Fig. 7. The air is compressed in the axial or centrifugal compressor A and heat is added at B through a heat exchanger associated with external combustion. The hot gas passes through the turbine D and the residual heat is removed by means of the heat exchanger F and transferred to the air leaving the compressor. The air leaving the heat exchanger recirculates and enters the compressor without further change in pressure and temperature. High efficiencies are obtainable with this cycle, as shown by curves of J. Ackeret in Fig. 8. It is seen from the figures that the level of efficiency occurs at relatively low values of pressure ratio, which means that a compressor-turbine unit with fewer stages could be used. The disadvantage of the system is the large amount of heat exchange surface required, which would tend toward a unit much too bulky and heavy for present consideration. However, the inherent advantage of good efficiency would indicate research on this cycle to be desirable. It is recommended that estimates be made of the over-all weight, size, and performance of this system. Another possibility which should be investigated is the utilization of different gases for this cycle, such as freon, carbon dioxide, argon, helium, etc.

10. *Displacement Compressor Units.*

There are some proposed gas-turbine units, such as the Lysholm type, which utilize a displacement type of rotary compressor in place of the more common axial or centrifugal. The displacement compressor had the advantage of high compression ratio,

but is essentially low in capacity. Therefore, in order to obtain the flow needed by gas turbine units, a system is required which involves weight and complication.

11. *Turbofan.**

The advantage of the turboprop over the turbojet in specific fuel consumption for subsonic flight speeds is due to the higher propulsive efficiency of the propeller. This is, in turn, due to the lowered velocity of the slip stream compared with the exhaust jet velocity of the turbojet. To improve the fuel economy of the turbojets, one can allow practically all the available power in the combustion gas from the combustion chamber to be converted into mechanical power by the gas turbine. The excess power over that required for driving the compressor is then used to drive a separate fan which compresses air from the atmosphere in a duct. The compressed air then mixes with the exhaust gas from the turbine and the mixture discharges through the tail pipe. This is the turbofan as shown in Fig. 9. Due to the increased air mass used, the exit momentum is larger than the turbojet with a corresponding reduction in the discharge velocity. The propulsive efficiency is thus increased and specific fuel consumption reduced.

For operations at transonic speeds, the turbofan has the added advantage over the turboprop in that the air duct will slow down the air stream before arriving at the fan. The relative velocity of the fan blade with respect to the air stream can be kept at a favorable value for high efficiency. On the other hand, conventional propellers of high efficiency at transonic speeds are difficult to design. British investigators have shown that with the present design of propellers, the turbofan should give greater fuel economy than the turboprop for flight velocities over 550 mph.

At supersonic flight velocities, the possibility of additional combustion in the tail pipe of the turbofan should also be considered. For such operations, the turbofan is not unlike the ramjet. In fact, if for a given air flow one can imagine a series of designs with gradually decreasing amounts of air going through the compressor-gas turbine system of diminishing size, then the ultimate power plant, with no air at all passing through the gas turbine, is the ramjet. In this sense then, the turbofan with tail-pipe burning is an intermediate between the turbojet and the ramjet. At supersonic speeds, the thrust of such a system will be approximately that of a ramjet but the weight will be slightly heavier than the ramjet. However, it will be much lighter than a turbojet of equal thrust since only part of the total air flow goes through the compressor gas turbine system, with the consequent reduction in the size of the gas-turbine. Furthermore, the turbofan with tail-pipe burning can also develop thrust at low speeds. It is, thus, a more versatile power plant than the ramjet. Therefore, further investigations of this power plant are recommended.

12. *Current Gas-Turbine Developments.*

Brief data on several existing and proposed gas-turbine units are given in Table II.

TURBOJET

1. *General Description.*

A schematic diagram of a typical turbojet unit is shown in Fig. 10. It is similar in principle to the gas-turbine cycle described in the previous chapter, except for the fact

* Contribution of Dr. H. S. Tsien to this section is acknowledged.

that there is no propeller drive, and all the net output of the unit goes into the jet. The turbine extracts just enough heat to drive the compressor. The work developed by the turbine, therefore, is equal to the work absorbed by the compressor. The remaining energy of the hot gases leaving the turbine goes into thrust upon expansion in the nozzle.

Calculations were made by M. Alperin of the specific fuel consumption and specific thrust for a series of assumed values of turbine-inlet temperatures, compressor efficiency, turbine efficiency, and pressure ratio.* With the aid of these studies, it is possible to draw general conclusions regarding the factors influencing the performance of turbojets, and to estimate the order of magnitude of the gain to be expected for various future improvements.

2. *Factors Influencing Performance.*

Since the thrust of a turbojet unit is obtained by expansion of hot gas through the tail pipe, it would be expected that higher compression ratio would increase the efficiency of the system. However, there are several factors which limit the validity of this statement. First, there are losses in the turbine and compressor, and these losses increase with compression ratio. Therefore, a point is eventually reached at which the increase of losses is greater than the theoretical gain. Also, it should be noted that the over-all efficiency in flight is equal to the product of thermal efficiency of the gas turbine unit alone times the propulsive efficiency. The propulsive efficiency, in turn, depends on the ratio of jet velocity to flight velocity. The factors influencing the performance of a turbojet are in general those influencing the gas turbine, as previously discussed.

3. *Influence of Speed and Altitude.*

Flight velocity influences the performance of a turbojet in various ways. The ram pressure increases at the compressor inlet with increasing speed, the mass flow increases, and the compressor and turbine reach a new equilibrium condition. The differential between jet speed and flight speed also changes. It is difficult at present to make general statements since for any specific unit the new equilibrium condition has to be computed for each new flight speed. Curves for a specific unit, the I-40, are shown in Figs. 11, 12, and 13.

With increasing altitude the temperature of the air entering the compressor inlet decreases. This is a favorable effect if the turbine-inlet temperature is maintained at a constant limiting value, conditioned by the metallurgy of the turbine blades. The lower compressor-inlet temperature, therefore, combined with the same turbine-inlet temperature, would mean an over-all increase in temperature ratio which is favorable for cycle efficiency. However, the Mach number relative to the compressor tip is increased due to the lower value of sound velocity, which usually results in lower compressor efficiency. Since the velocities in the unit remain approximately constant and independent of altitude, it would be expected that the mass flow would decrease approximately in proportion to the density. The thrust, in turn, would decrease with the mass flow. The influence of altitude on the performance of the I-40 is also illustrated in Figs. 11, 12, and 13.

* The Research Branch of the Power Plant Laboratory, Wright Field, has made similar calculations, limited, however, to the region of subsonic flight.

4. *Methods of Obtaining Temporary Increase of Thrust.*

Methods of obtaining temporary increase of thrust are especially important for jet engines at both ends of the speed range. For take-off the over-all efficiency of the jet unit is low due to the high residual knietic energy associated with the large differential between jet velocity and airplane speed. Since the efficiency of the turbojet increases with rpm, cruising conditions are closer to the maximum than is the case for a conventional engine. Increased thrust is, therefore, desirable for temporary increase of speed.

One method of obtaining a temporary increase of thrust is by afterburning or supplementary combustion in the tail pipe. The amount of thrust increases with increasing pressure ratio. Therefore, afterburning is especially effective with high pressure ratio gas turbines, and the thrust increases rapidly with flight speed due to the ram pressure. In the report by Sears, Ashkenas, and Hasert, it was estimated that although supersonic speeds were questionable for turbojets now in operation, afterburning would furnish enough thrust to make supersonic speeds appear feasible. Due to the importance of this question, it is recommended that systematic theoretical and experimental studies be conducted to explore the possibilities of afterburning, especially at supersonic speeds.

Another promising means of obtaining increased thrust is by liquid injection, especially in combination with afterburning. Injection of ammonia and fuels should receive further attention, especially with regard to oxygen-bearing fuels for altitude operation.

5. *Estimates at Supersonic Speed.*

Experimental data on the performance of turbojet units above the speed of sound are greatly needed at the present time for the predictions of future application of supersonic aircraft. Some theoretical estimates of turbojet performance at supersonic speeds have been made by M. Alperin. The assumption was made that a shock wave forms in front of the duct entrance of the turbojet, and that the flow through the turbojet is subsonic. Pressure, density, and temperature changes across the shock wave were calculated by the usual shock equations. With these new values of pressure, density, and temperature at the inlet, the turbojet performance was calculated in the normal way. Sample curves showing the influence of pressure ratio on specific fuel consumption for several Mach numbers are given in Fig. 14. It is to be noted that the optimum pressure ratio of the turbojet compressor decreases with increasing Mach number as would be expected because of the increasing contribution of ram pressure. It is seen from the curves that as Mach number increases a value will be reached where the optimum compressor pressure ratio is one, and the turbojet becomes a ramjet. The transition point between turbojet and ramjet depends on the assumptions as to turbine-inlet temperature, compressor efficiency and turbine efficiency. The higher these values the higher the Mach number at which a turbojet will be effective.

The influence of turbine-inlet temperature on specific fuel consumption and specific thrust is shown in Figs. 15 and 16 respectively, for a flight Mach number of 1.6.

It should not be concluded from the above, however, that turbojet units now in operation are capable of propulsion at supersonic speeds. In the report by Sears,

Ashkenas, and Hasert, it is shown that using scaled-up test results of the thrust per unit frontal area for the Westinghouse 9.5A turbojet unit, the drag of the frontal area would become equal to the thrust at about 800 mph, so that the unit could not propel its own frontal area above this pseed unless afterburning were employed. However, estimated thrust values of the Lochkeed L-1000 appear sufficient for supersonic flight, without afterburning, and Lockheed estimates with afterburning show a 100% increase of thrust. The curves shown in Fig. 16 emphasize the fact that a very considerable increase of thrust for the same frontal area could be obtained if applied research can develop methods of increasing turbine temperature limits. In addition, the utilization of afterburning at supersonic speeds should be intensively investigated since, even with present-day units, the increase of thrust should be sufficient to make supersonic flight possible.

Internal duct flow for bodies traveling at high subsonic or supersonic speeds is a relatively unexplored field. Tests by the NACA have shown promise of developing effecient supersonic diffusers. Considerable applied research in this field is needed. It appears possible to develop diffusers which, instead of having a strong shock in front of the duct entrance have a weaker shock inside the duct, thereby decreasing the loss. Control of this shock by boundary layer suction should be further investigated. Cone entry diffusers appear especially promising.

6. *Combustion.*

The combustion process opens a new field for analysis in that both thermodynamics and aerodynamics must be taken into consideration. This forms, in fact, a new science which has recently been referred to as aerothermodynamics. The fuel is sprayed into the combustion cylinder through a series of spray nozzles, and combustion takes place in the presence of a steady flow of air. Since the amount of fuel is small in proportion to the air, the process involves an addition of heat content without essential change of mass flow. The equations are, therefore, similar in nature to those for a shock wave. If conditions upstream of the combustion process are denoted by a subscript 1, and conditions downstream by 2, and H represents the heat added by the combustion process per unit mass flow, then the three basic equations for mean changes through the combustion chamber are:

$$\rho_1 V_1 = \rho_2 V_2 \qquad \text{Continuity of mass} \qquad (1)$$

$$P_1 \rho_1 V_1^2 = P_2 + \rho_2 V_2^2 \qquad \text{Continuity of moment impulse} \qquad (2)$$

$$\frac{k\text{-}1}{k}\frac{P_1}{\rho_1} + \frac{V_1^2}{2} H = \frac{k}{k\text{-}1}\frac{P_2}{\rho_2} + \frac{V_2^2}{2} \qquad \text{Conservation of energy} \qquad (3)$$

For a given set of entering conditions the amount of heat added can be plotted as a function of either pressure or Mach number at the downstream section. The pressure decreases in a parabolic function as the amount of heat is increased. However, the significant fact emerging from the analysis is that there is a definite maximum to the amount of heat that can be added. The equations show that the maximum added heat corresponds to a discharge Mach number M_2 equal to 1. For this maximum condition of added heat, a curve can be plotted of maximum heat versus inlet Mach number. This

is shown in Fig, 17. It is noted that the maximum heat which can be added increases rapidly with decreased inlet Mach number. This would indicate that low entrance velocities are desirable from the point of view of adding as much heat as possible. However, it should be pointed out that lower velocities are associated with larger combustion chambers which mean larger and heavier units, so that for this particular item a compromise must be made. One of the chief problems in combustion is turbulent mixing and diffusion to obtain an efficient and uniform combustion process. There has been considerable aerodynamic research on the turbulent flow of air alone, but there is a need for research on the turbulent mixture of air and fuel droplets. This is especially true at altitudes where the air density is very low.

One of the difficulties of present units is the nonuniformity of the combustion process. The distribution of temperatures around an annular ring is irregular and high points or hot spots occur. This means that the mean turbulent temperature has to be kept at a lower value than would be the case if the temperature were uniform and regions of higher temperature did not exist.

Combustion at altitude is a problem about which there is very little known. Various factors become critical at altitude, for instance, the air becomes too rare for proper diffusion and mixture; the spark becomes too weak for proper ignition; the distance between the fuel nozzles and the point of ignition may be wrong; the compressor and turbine may be far enough off their design operating point to cause pumping and instability; the relative Mach number at the compressor inlet increases with the decreasing temperature, which may lead to increased losses and choking. All these subjects should be the subject of intensive investigation in altitude combustion laboratories and in high-speed, altitude wind tunnels.

7. Utilization of Nuclear Energy.

It is expected that the application of nuclear energy will revolutionize propulsive methods, but it is too early to make an engineering evaluation of its possibilities. Some speculation as to the application of nuclear energy to propulsion is given in a report by H. S. Tsien.

In general, it would appear that the application of nuclear energy to gas-turbine propulsion would remove the item of specific fuel consumption as a factor of consideration, and range would automatically become unlimited. The chief question is one of initial weight and the utilization of the energy at the proper rate. For supersonic turbojet application, for instance, the use of nuclear energy by itself might not contribute directly to enable the turbojet to fly faster than sound. This would still be a problem of providing greater thrust per unit frontal area, which is bound up with the basic turbojet development problems, such as increasing turbine-temperature limits and improving compressor and turbine efficiencies. However, the item to be pointed out very strongly is that if applied research can develop improvements in basic turbojet performance to make the turbojet capable of supersonic flight, then with the utilization of nuclear energy the range at supersonic flight becomes infinite. The possibility of future applications of nuclear energy, therefore, appears to emphasize the importance of intensive development of gas turbine components.

8. *Existing and Proposed Turbojet Units.*

Several of the more representative types of existing and proposed turbojet units are presented in Table III for American units, Table IV for British, Table V for German, and Table VI for Japanese units.

RECOMMENDATIONS FOR FUTURE RESEARCH

1. *Limitations of Propellers for High-Speed Flight.*

In the various studies of the relative performance of reciprocating engines plus propeller drive, gas turbine plus propeller drive, and turbojet, the main factor which hinders the comparison is lack of information on propeller efficiency at speeds above 500 mph. Estimates of high-speed propeller performance are usually based on two-dimensional airfoil data. However, recent altitude flight tests indicate that the three-dimensional flow associated with an actual rotating propeller may not be as critical as two-dimensional blade theory would predict. In addition, German wind-tunnel tests on propellers with swept-back blades indicated improved efficiency at transonic relative speeds near sonic. Therefore, it is recommended that test data be obtained as soon as possible in high-speed wind tunnels and in flight on the behavior of conventional and swept-back propellers at transonic and supersonic speeds; also, that development and testing of propeller sections specifically designed for supersonic speeds be expedited.

2. *Interrelationship Among Size, Weight, Thrust, Pressure Ratio, Turbine-Inlet Temperature, Specific Fuel Consumption.*

As discussed in previous sections of this report, it is evident that the performance of a gas-turbine cycle is influenced greatly by the selection of pressure ratio, turbine-inlet temperature, etc. However, theoretical advantages, as shown by such analysis, must be weighed against the practical aspects such as size, weight, life, simplicity, and maintenance. For instance, if the analysis for a certain application shows a gain in fuel economy for an increased pressure ratio, it should be kept in mind that the higher pressure would normally be obtained by increasing the number of stages. This would add to the rotor weight which, in turn, would involve other weight increases in the unit. Whereas this increase might not be justified for high-speed applications at short range, it might well be profitable for long-range cruising. The danger of making general conclusions on the basis of specific cases should be avoided, and all aspects reconsidered for each new application.

3. *Lighter Weight by Improved Construction and Materials.*

The development of gas-turbine and turbojet units to date has shown that there is still room ahead for improvements in the specific weight, or ratio of weight to thrust. It is to be noted, for instance, that in the successive development of five different Westinghouse turbojets there has been a consistent decrease of specific weight from unit to unit as follows: .66 - .56 - .49 - .44 - .36. The British RB-41 has a specific weight of .31. Manufacturers believe it possible to attain a value of .25 pound per pound of thrust.

Further material research should be conducted on alloys such as beryllium, aluminum, magnesium, etc. Parts such as casings, etc., might make use of ceramic sandwich construction with resin binder.

4. *Compressor Research.*

Axial compressors have been developed to a fairly high state of efficiency by using many stages of relatively highly loaded airfoil-type blading. The use of lattice-type blading with closely spaced and highly cambered blades, as developed at Wright Field for supersonic wind-tunnel compressors, would greatly increase the stage pressure rise, and, therefore, decrease the number of stages. This would be quite advantageous towards decreasing the weight and complication of a gas-turbine unit providing high efficiency could be maintained. However, the chief factor limiting the development of higher stage pressure ratio is the almost complete lack of experimental data on the behavior of blade sections in lattice arrangement at high Mach number. Since the stage pressure developed by a compressor is approximately proportional to the square of the inlet Mach number, it is apparent that any gain in inlet Mach number is very advantageous provided that no excessive loss of efficiency is involved. High-speed blade research should be actively sponsored not only in two-dimensional stationary lattice models, but also on three-dimensional rotating grids.

Boundary layer control applied to compressor blades has been considered by the Power Plant Laboratory, Wright Field, but further investigation is needed to see if the increased complication is justified.

Another possibility is the utilization of supersonic flow in the compressor. Encke of Göttingen experimented with wedge-shaped rotor blades at supersonic relative velocities, but found the operating condition too far away from the design point to warrant any conclusion to be drawn. Weise of the DVL has an experimental compressor whereby the velocity leaving the rotor blades was supersonic. The stator vanes behind the rotor were arranged to cause a normal shock which added to the pressure rise. Estimates showed promise of considerable increase in stage-pressure rise in this way, but preliminary experiments showed deviation from the design condition too great to be conclusive.

Considerable increase in mass flow and associated thrust per unit frontal area of a gas-turbine unit is believed possible by axial compressor research and development. Wind-tunnel compressor design at Wright Field has indicated that smaller hubs might be used than is usually used in present gas-turbine systems. This means increased disk area and associated increase mass flow and thrust. At the same time it is believed that the axial velocity components relative to the blades can be increased, with still further increase of mass flow and thrust. To explore these possibilities requires intense applied research in the field of compressor design, especially for three-dimensional flow in rotating systems.

It is recommended that the subject of high-speed flow in rotating machinery be given much more attention in the future. High-speed component test facilities are needed.

5. *High-Temperature Materials.*

The development of high-temperature materials is especially important in view of the considerable increase of thrust or power per unit frontal area which occurs with increase of the turbine-inlet temperature. It should be emphasized that the fatigue characteristics at high temperatures are of greater importance than simple rupture

strength. The development of ceramics should be given special attention because they give promise of significant temperature increase. Research on the development of ceramic alloys which can be used as the construction material itself should be expedited.

6. *Turbine Research.*

Aerodynamic research on turbines is needed on problems of three-dimensional flow, degree of reaction, influence of tip clearance and interaction with nozzle flow. The present turbojet units tend to use impulse-type of turbine blading because it is possible to utilize only one stage for the lower pressure ratios of such units as the I-40 and TG-180. Reaction blading appears to be somewhat more efficient, but since it has less pressure drop, more stages are required. There is an important question, therefore, as to the interrelationship between efficiency and the number of stages. Aerodynamic factors contributing to the forcing function for turbine-blade vibration should be investigated. Utilization of hollow blades, especially in conjunction with blade cooling or boundary layer suction, is an important research problem, in the light of utilizing higher gas temperatures with associated thrust increase.

With respect to the cooling of turbine blades several German projects are of interest. One project by Schmidt of the LFA was concerned with water cooling of the blades. The steam resulting from the heating of the water was utilized in a steam turbine which could either feed back into the driveshaft or be used to furnish auxiliary power for the aircraft.

Another similar project was by Rietz of Göttingen who was experimenting with sodium instead of water as the cooling medium.

Cooling of walls by seepage through porous material as mentioned in.the report by Duwez should be investigated for application to turbine blades.

In general, research on blade cooling is recomended because of the possibilities of increased thrust per unit frontal area. The cost in weight and complication should be estimated throughout in order to have realistic evaluations.

7. *Combustion Research.*

This subject is considered especially important in view of the primitive state of knowledge regarding even the fundamental processes of combustion. Investigation should be made as to the physical principles at work and the factors of similarity which make it possible to predict behavior in the different operating conditions. Especially important are questions of stability at altitude and behavior over wide operating ranges. For flight above the speed of sound, the effect of flow changes through the jet unit should be investigated as to their effect on combustion. The relative merits of the annular type of combustion chamber versus the individual type should be studied Interests of size and weight dictate small, compact combustion chambers. This, however, involves higher air speeds through the combustion chamber and sufficient investigation should be made to enable reasonable design compromises to be reached The mixing of fuel with air should be studied, especially under altitude conditions where the air density is low. Also in dealing with combustion chambers, the question of safety becomes important, and the construction should be considered from this viewpoint.

8. *Fuel Research.*

It is recommended that more study be given to the heat content per unit volume of various fuels as well as the heat content per pound, since with the large quantities of fuel required by jet airplanes, the frontal area is a significant factor. Combustion efficiency should also be studied, since a fuel with a high energy content may lose its advantage if the combustion efficiency is low. Hydrogen is attractive from a heat value standpoint and investigation is recommended as to the possibility for its utilization.

Development of fuels for high-altitude operation: It may be necessary to carry fuel with a certain percentage of self-oxidizer to be used for altitudes above the limits of normal combustion.

Of paramount importance for the future is the possible utilization of atomic fuels. This is discussed in the report by H. S. Tsien. It is pointed out that although the heat content of atomic fuels range between 3×10^8 and 3×10^{10} BTU/lb as compared with 1.87×10^4 BTU/lb for gasoline, the significant factor for its utilization in gas turbines, etc., is the rate of energy release, which is about 5×10^4 BTU/hr/lb, for a typical case. Although low, application to gas-turbine propulsion appears within the range of feasible application.

There is a distinct need for engineering development work to apply the abstract results of nuclear physics to engineering utilization of nuclear power for propulsion.

9. *Cycle Research.*

It has been noted in an earlier part of this report that considerable gain in fuel economy can be obtained by use of regeneration, reheating, use of a closed cycle with gas under pressure, etc. However, the advantages are counterbalanced by increased weight and complication. It would be highly important to know to what degree weight and complication might be cut down by suitable research on heat exchangers and other such items entering these cycles.

It is recommended that the component parts involved in cycle improvement be subjected to systematic research regarding future improvements. Similar studies should be made on the compound engine, the free-piston system, the turbofan, and the use of new gases in csnjunction with a closed cycle.

10. *Duct Research.*

At high speeds the aerodynamics of shapes at entrance and exits of ducts, as well as the interior portions, become increasingly important, especially at transonic and supersonic speeds. It has been found for instance, that an increase in diffuser efficiency at supersonic speeds is obtainable by cone, entrances, and also by suitable boundary layer control. The whole subject of high-speed flow in ducts should be the subject of combined theoretical and experimental research. This should also include the effect of angles of attack or yaw on the flow, since unbalanced entrance condition can ruin the aerodynamic performance of compressors or turbines, and if the nonuniformity is great enough it may lead to blade failures. Theoretical work on compressible flow should also include the flow through both the rotating and stationary blades of compressor and turbine.

In this connection also, it is important to study means of varying the flow conditions in the duct to suit different speed or altitude requirements. This is especially true when supersonic speeds are obtained in the tail pipe since only one speed of air passing through the duct can be obtained for any one duct shape. Also, the use of variable vanes for the compressor or turbine should be studied since one of the limitations on compressor and turbine design at present is the necessity for having stable operation over a wide range. In general, the interrelationship between turbine and compressor characteristics should be investigated so as to eliminate danger of stalling or pumping under certain flight conditions.

11. *Increased Thrust for Take-Off and High Speed.*

The providing of increased thrust for take-off is of great importance for jet units since take-off is a critical condition. Liquid injection, afterburning, etc., should be the subject of systematic research. Since the increased thrust due to afterburning is considerably increased by ram, it becomes of significant magnitude at high speeds, and should play an important part in obtaining supersonic speeds.

Other miscellaneous research problems are: the development of lighter and more convenient means of starting jet units, including self-starting devices; lubrication research, especially for larger units operating over wider range of thrust, temperature, altitude, etc.; determination of optimum clearances; dirt and erosion problems; icing problems; and the development of cheap expendable units for pilotless aircraft.

TABLE I — RECIPROCATING ENGINES

Manufacturer	Wright Aeronautical	Pratt & Whitney	Lycoming
Model	R-3350-57	R-4360-27	XR-7755-3
Type	18-Cyl Single-Stage Single-Speed Supercharger	28-Cyl Single-Stage Variable-Speed Supercharger	36-Cyl Dual Rotation (Two-Speed Reo Gear) Fuel Injection Single-Stage Single-Speed Supercharger
Sea Level Performance	Take-Off 2200 hp 60% Cruise 1200 hp	Take-Off 3000 hp 60% Cruise 1500 hp	Take-Off 5000 hp Medium Cruise 2000 hp
Speed RPM	2800 rpm 2000 rpm	2700 rpm 2144 rpm	2600 rpm 1600 rpm
Air Flow	4.45 lb/sec 2.22 lb/sec	6.11 lb/sec 3.13 lb/sec	
Specific Fuel Consumption	.80 lb/hp-hr .44 lb/hp-hr	.73 lb/hp-hr .46 lb/hp-hr	.70 lb/hp-hr .37 lb/hp-hr
Compression Ratio	6.85:1	7:1	8.5:1
Over-all Length	76.26 in.	96.75 in.	121 in.
Over-all Diameter	55.94 in.	52.5 in.	61 in.
Total Dry Weight	2758 lb	3404 lb	6050 lb

TABLE II — GAS TURBINES (American Types)

Manufacturer	G. E.	Wright	Northrup	Frederick Flader	Westinghouse
Model	T-31 (TG-100)	T-35-1	T-37-1	XT-33-1	X25D2
Diameter (in.)	37	59		42	33
Length (in.)	115-3/4	140		192	173-1/2
Weight (lb)	1950	3800	4000	5000	2540
Power (hp)	2190 bhp + 625 lb thrust Sea Level 0 mph 2780 bhp +340 net jet hp Sea Level 410 mph	5000 bhp +1300 lb thrust Sea Level 0 mph 5000 est hp at 20,000 ft at 500 mph	4000 bhp at 35,000 ft at 400 mph	4000 est hp at 35,000 ft at 500 mph	3300 shaft hp +980-lb thrust Sea Level 0 mph
RPM	13,000	7080	6250	6000	12,400
Specific Fuel Consumption (lb/bhp-hr)	0.81 Sea Level 0 mph 0.61 Sea Level 410mph 0.52 at 35,300 ft 410 mph	0.90 Sea Level 0 mph 0.58 at 20,000 ft at 500 mph	0.45 at 35,000 ft at 400 mph	0.50 at 35,000 ft at 500 mph	0.615
Mass Flow (lb/sec)	22	73	32 at 35,000 ft at 400 mph		40
Type Compressor	14-Stage Axial	2-Stage Centrifugal	14-Stage Axial	Axial	Axial

NOTE: These data furnished by Power Plant Laboratory, Wright Field, based on information available as of 2 November 1945.

TABLE III — TURBOJET UNITS (American Types)

Manufacturer	G. E.	G. E.	G. E.	Westing-house	Westing-house	Westing-house	Lockheed
Model	J-35 (TG-180)	J-33 (I-40)	J-31 (I-16)	19XB-2A	9.5A	24C	XJ-37-1 (L-1000)
Diameter (in.)	38	50.5	41.5	25-5/8	9-1/2	32 x 38	25
Length (in.)	177	102-7/8	72	93-9/16	52-1/2	112-9/16	123.5
Weight (lb)	2400	1920	885	826	140	1075	1235
Thrust (lb)	4000	3750	1550	1600	275	3000	5100 Take-Off (After-burning)
RPM	7600	11,500	16,500	17,000	34,000	12,000	15,600
Specific Fuel Consumption (lb/lb-thrust-hr)	1.08	1.20	1.25	1.144	1.50	1.066	1.76 at Sea Level Take-Off (After-burning) 1.07 at 600 mph
Mass Flow (lb/sec)	73	73.3	32.8	29.0	6.5	52	34
Type Compressor	11-Stage Axial	1 Double Entry Centrif-ugal	1 Double Entry Centrif-ugal	10-Stage Axial	6-Stage Axial	10-Stage Axial	16-Stage Inlet 16-Stage High Pressure Axial

TABLE IV — TURBOJET UNITS (British Types)

Manufacturer	de Havil-land	de Havil-land	de Havil-land	Armstrong Siddeley	Metropoli-tan Vickers	Rolls Royce	Power Jets
Model	Goblin III	Goblin IV	Ghost II	ASX-1 (a)	F2	RB-41 (b)	W2/500
Diameter (in.)	49.5	49.5	53	42	33	49.5	41
Length (in.)	93.5	93.5		83	109	96.8	
Weight (lb)	1530	1550	1950	1900	1500	1550	830
Thrust (lb)	33-3500	38-4000	5000	2600	1800	4070	1650
RPM	10,500	11,500	10,000	8000	7500	12,000	16,500
Specific Fuel Consumption (lb/lb thrust-hr)	1.15	1.08	1.08	0.96	1.04	1.03	1.14
Mass Flow (lb/sec)			82	50	47.5	80	34.7
No. Cumbustion Chamber	16 Straight Through	8 Straight Through	14 Straight Through	11 Straight Through	1 Annular	9 Straight Through	10 Reverse Flow
Type Compressor	Single Entry Centrif-ugal	Single Entry Centrif-ugal		14-Stage Axial	9-Stage Axial	Double Entry Centrif-ugal	Double Entry Centrif-ugal
Type Turbine	1-Stage	1-Stage		2-Stage	2-Stage	1-Stage	1-Stage
Pressure Ratio				5.0	3.5	4.0	4.1

(a) ASX-1—Unit has air inlet between compressor and turbine. Air flows forward and turns 180° through combustion chambers to turbine.

(b) RB-41—Run at 5000-lb thrust also, minimum Standard Fuel Consumption — 1.0 maximum — 1.05.

NOTE: These data furnished by Power Plant Laboratory, Wright Field, based on information available as of March 1945.

TABLE V

	TURBOJET UNITS (German Types)					TURBOPROP (German Types)	
Manufacturer	BMW	Junkers	Heinkel Hirth	Junkers	BMW	Daimler Benz	BMW
Model	109-003	109-004	109-011	109-012	109-018	021	028
Diameter (in.)	27	35	32	43	49	32	49.3
Length (in.)	124	152	99	194	165	138	228
Weight (lb)	1340	1640	1980	3520	5060	2560	7270
Thrust (lb)	1760	1980-2200	2860	6100	7500	Sea Level 0 mph 1950 hp + 1100-lb thrust Sea Level 500 mph 2750 est hp	Sea Level 0 mph 4700 hp + 4850-lb thrust Sea Level 500 mph 12,600 est hp
RPM	9500	8700	10,200		6000		
Specific Fuel Consumption	1.4 lb/lb thrust-hr	1.35 lb/lb thrust-hr	1.38 lb/lb thrust-hr	1.2 lb/lb thrust-hr	1.1-1.3 lb/lb thrust-hr	0.81 lb/hp-hr	0.58-0.68 lb/hp-hr
Type Compressor	7-stage Axial	8-stage Axial	1 Single Entry Centrif-ugal + 4-stage Axial	11-stage Axial	12-stage Axial	9-stage Axial	
Type Turbine	1-stage	1-stage	2-stage	2-stage	3-stage		

TABLE VI

	TURBOJET UNITS (Japanese Types)				MOTORJET (Japanese Type)
Manufacturer	Japanese Navy	Ishakawa Jima Shibaura	Nakajima	Mitsubishi	Hitachi
Model	NE 20	NE 130	NE 230	NE 330	TSU-11
Diameter (in.)	24.4	33.5	36 x 30	46.5 x 34.5	25.2
Length (in.)	106	152	135	157	86.5
Weight (lb)	990	1980	1920	2640	440
Thrust (lb)	1100	1980	1950	2860	440
RPM	11000	9000	8100	7600	9000
Specific Fuel Consumption (lb/lb-thrust-hr)	1.3	1.3	1.3	1.4	

NOTE: This information obtained by interrogation in Japan.

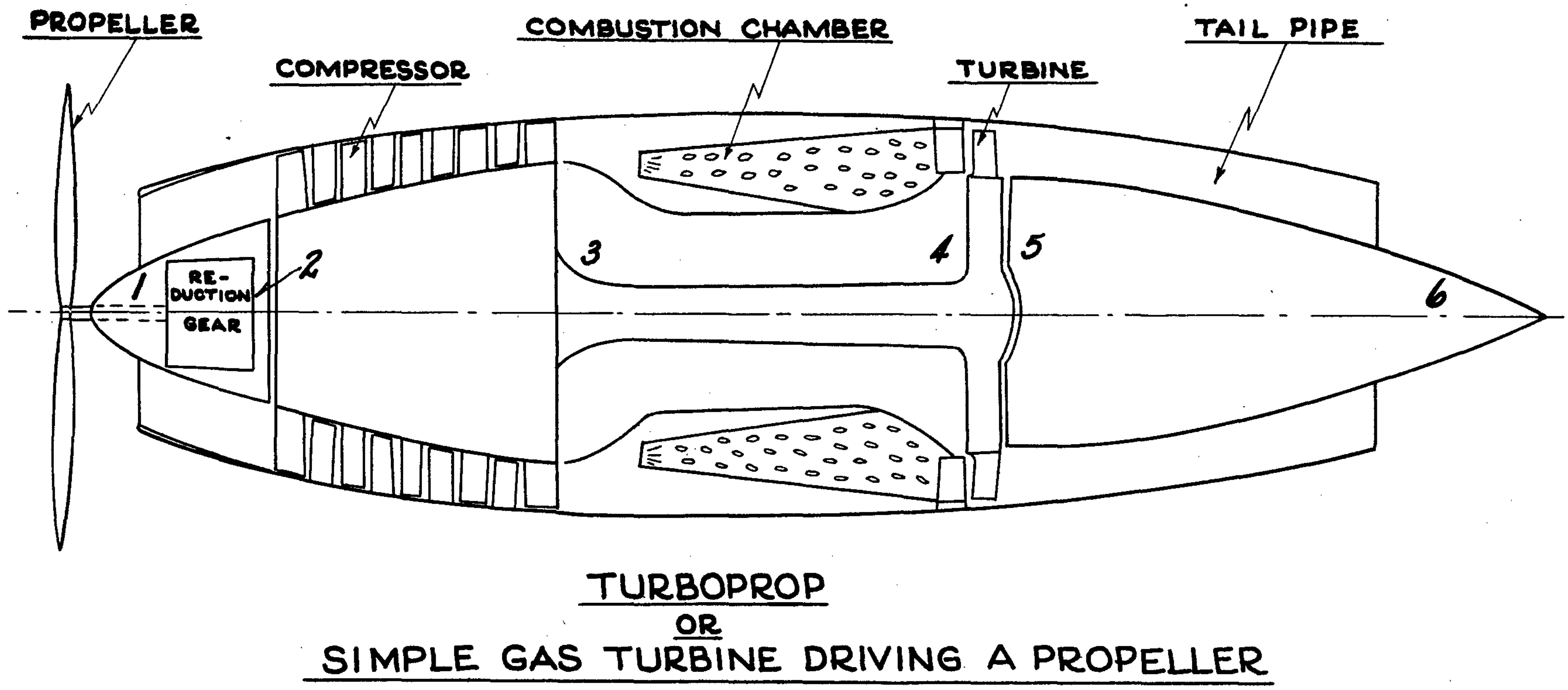


Figure 1 — Turboprop or Simple Gas Turbine Driving a Propellor

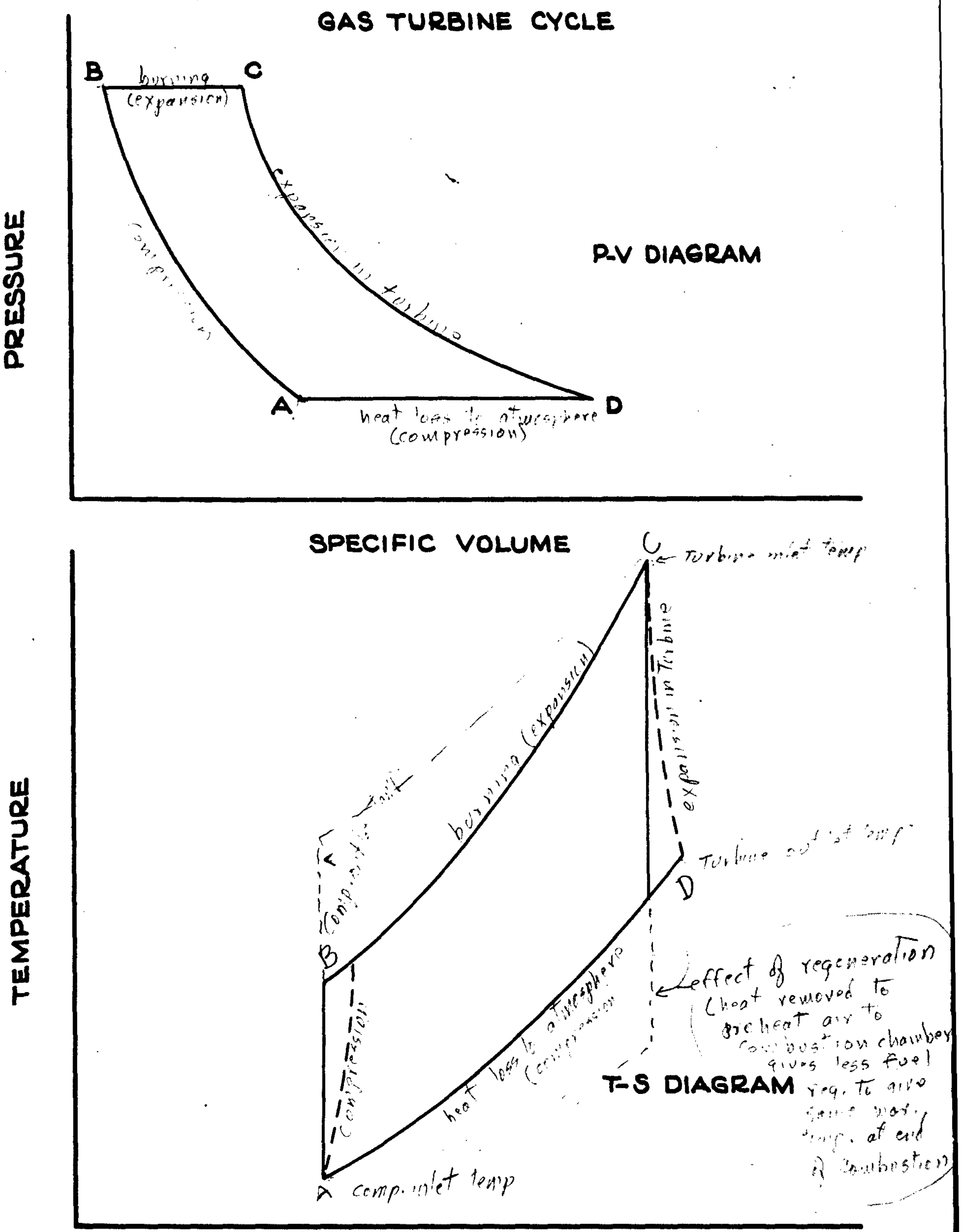


Figure 2 — (S) Entropy

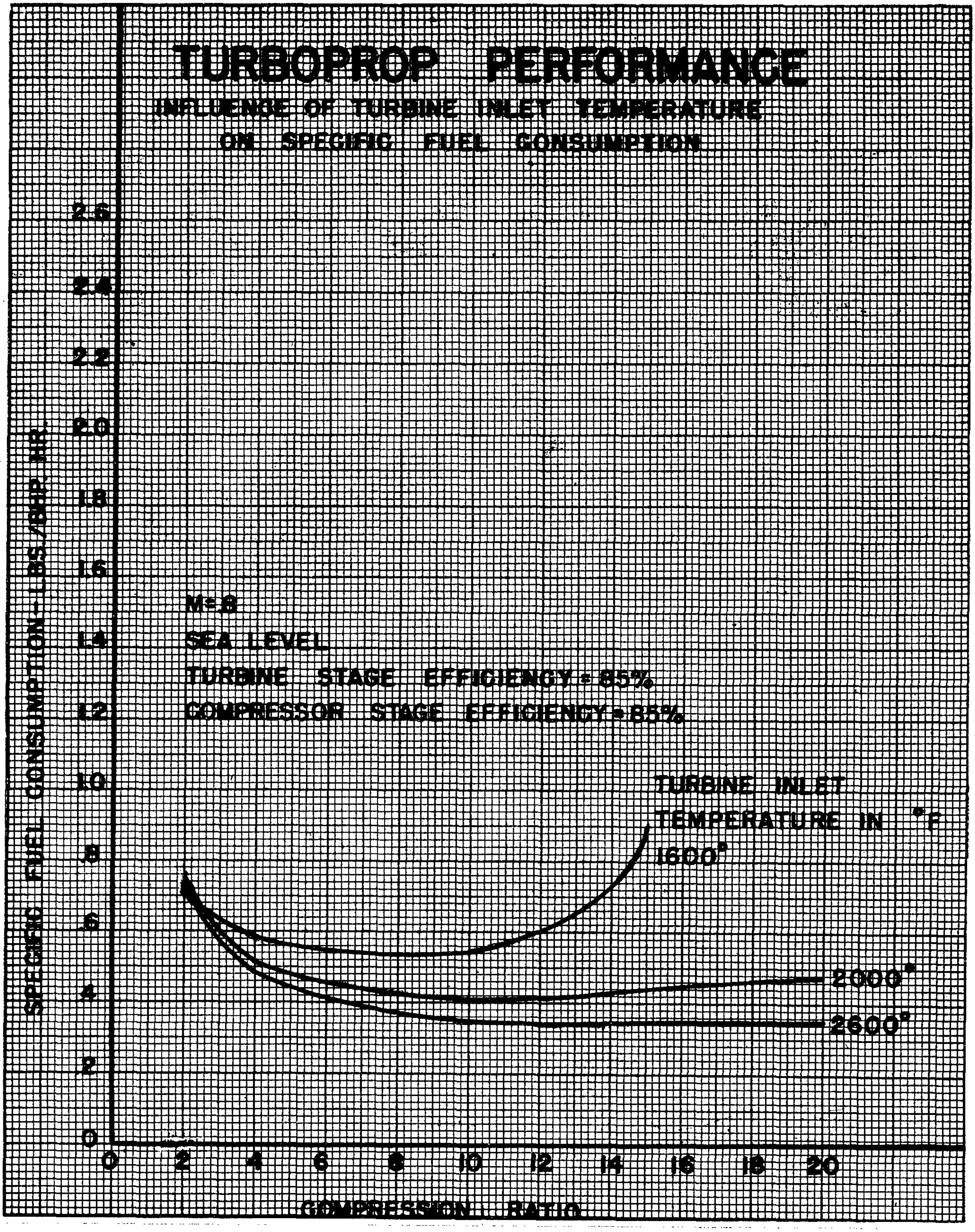


Figure 3 — Torboprop Performance (Influence of Turbine-Inlet Temperature on Specific Fuel Consumption)

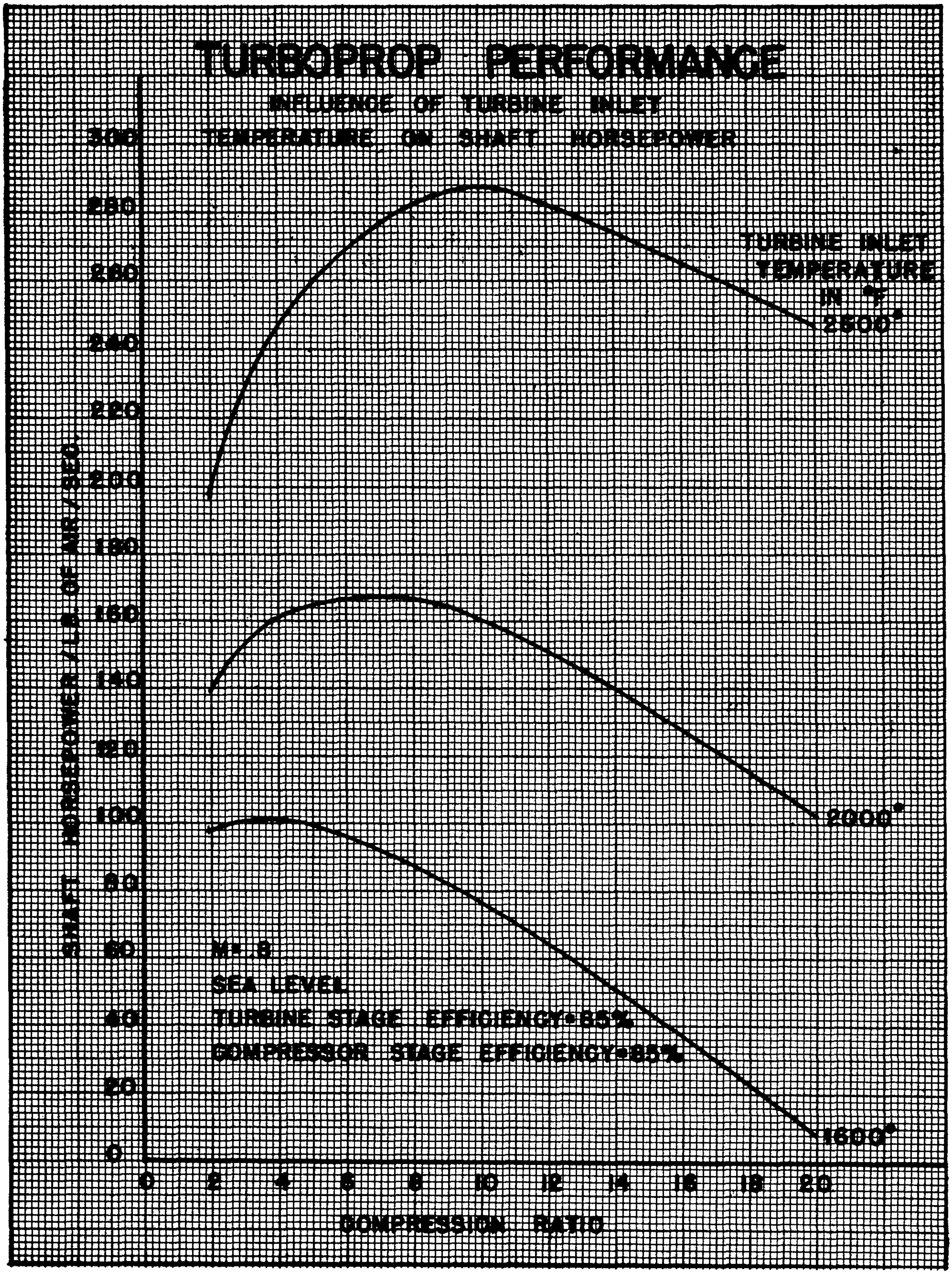


Figure 4 — Turboprop Performance (Influence of Turbine-Inlet Temperature on Shaft Horsepower)

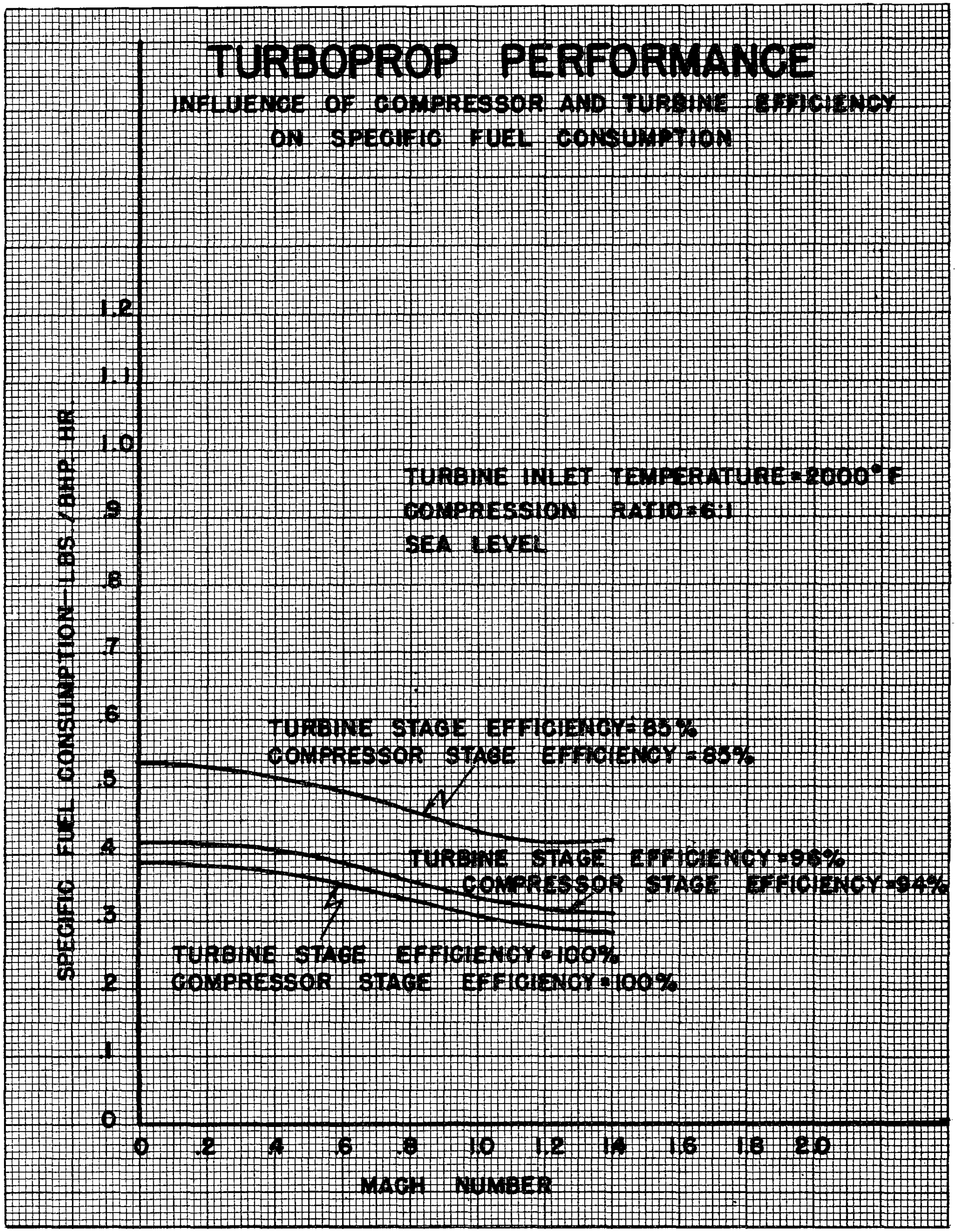


Figure 5 — Turboprop Performance (Influence of Compressor and Turbine Efficiency on Specific Fuel Consumption)

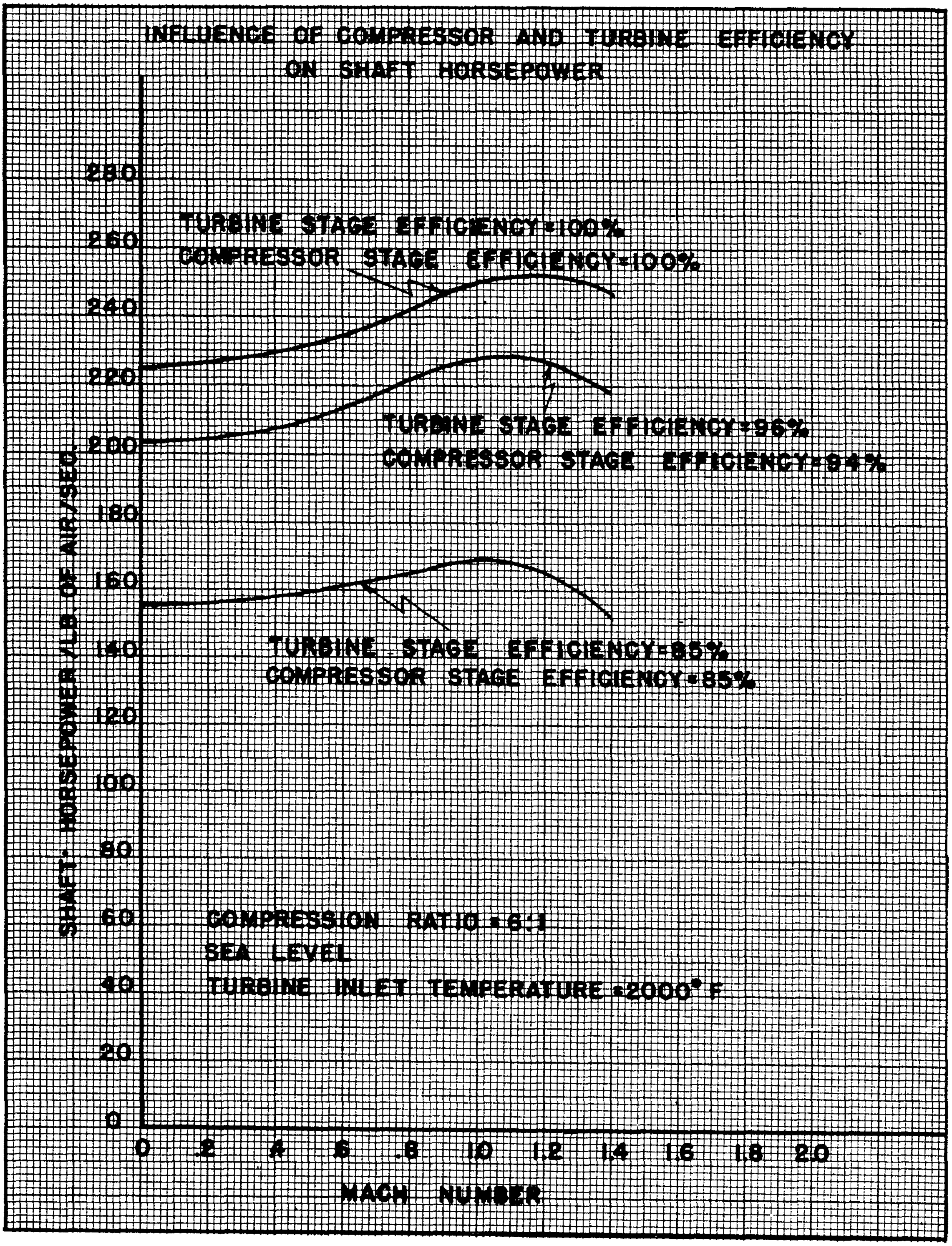


Figure 6 — Turboprop Performance (Influence of Compressor and Turbine Efficiency on Shaft Horsepower)

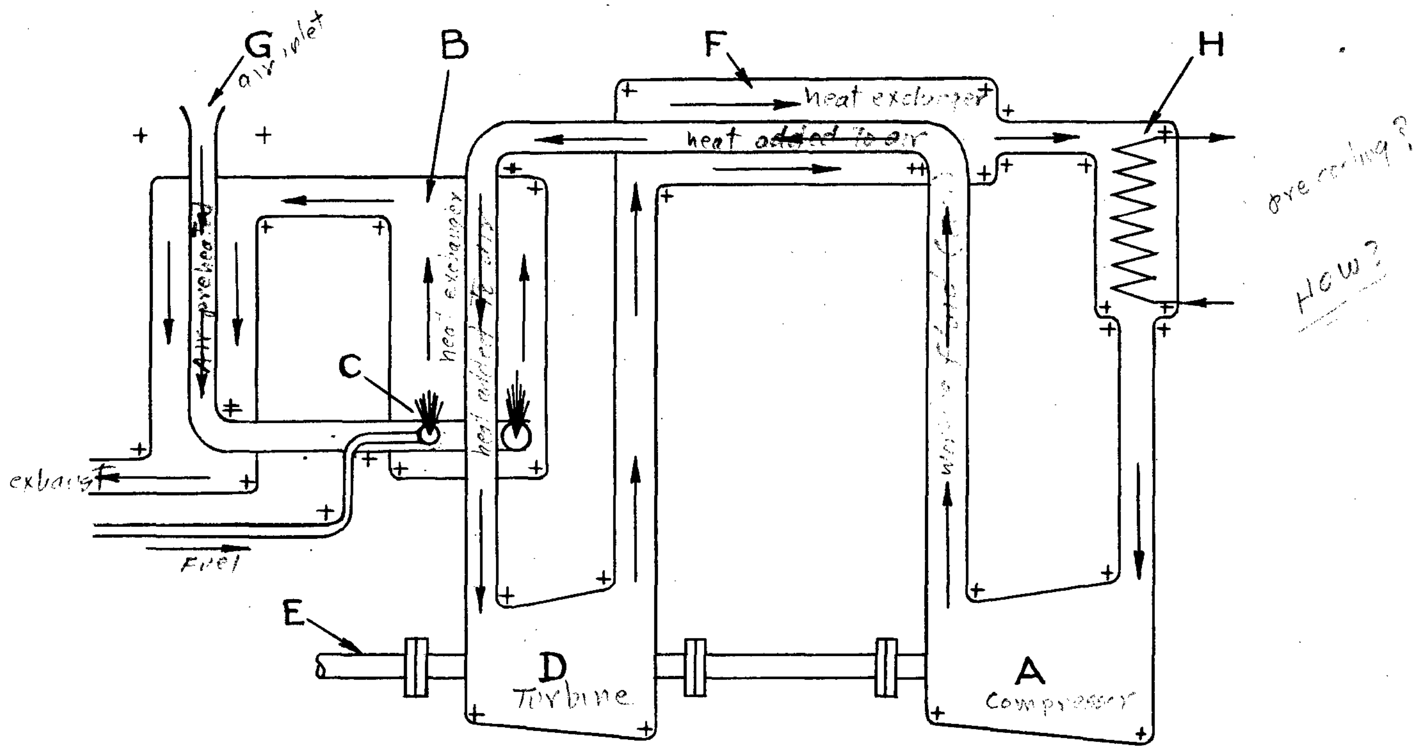
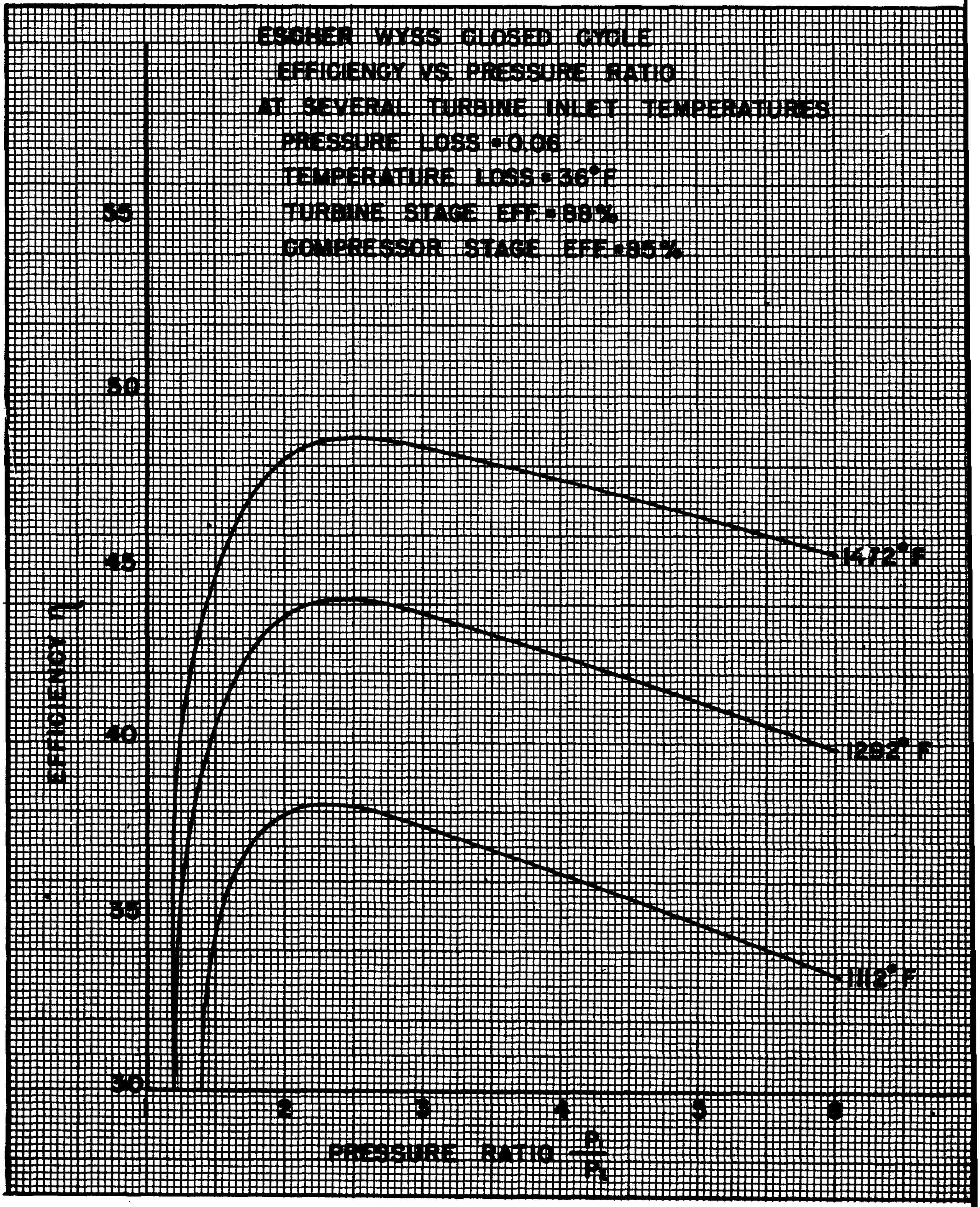


Figure 7 — Schematic Diagram of Closed Cycle Gas-Turbine System



ESCHER WYSS CLOSED CYCLE
EFFICIENCY VS. PRESSURE RATIO
AT SEVERAL TURBINE INLET TEMPERATURES
PRESSURE LOSS = 0.06
TEMPERATURE LOSS = 36°F
TURBINE STAGE EFF. = 88%
COMPRESSOR STAGE EFF. = 85%
55
50
45
40
35
30
EFFICIENCY η
1472°F
1292°F
1112°F
2
3
4
5
6
PRESSURE RATIO $\frac{P_2}{P_1}$

Figure 8

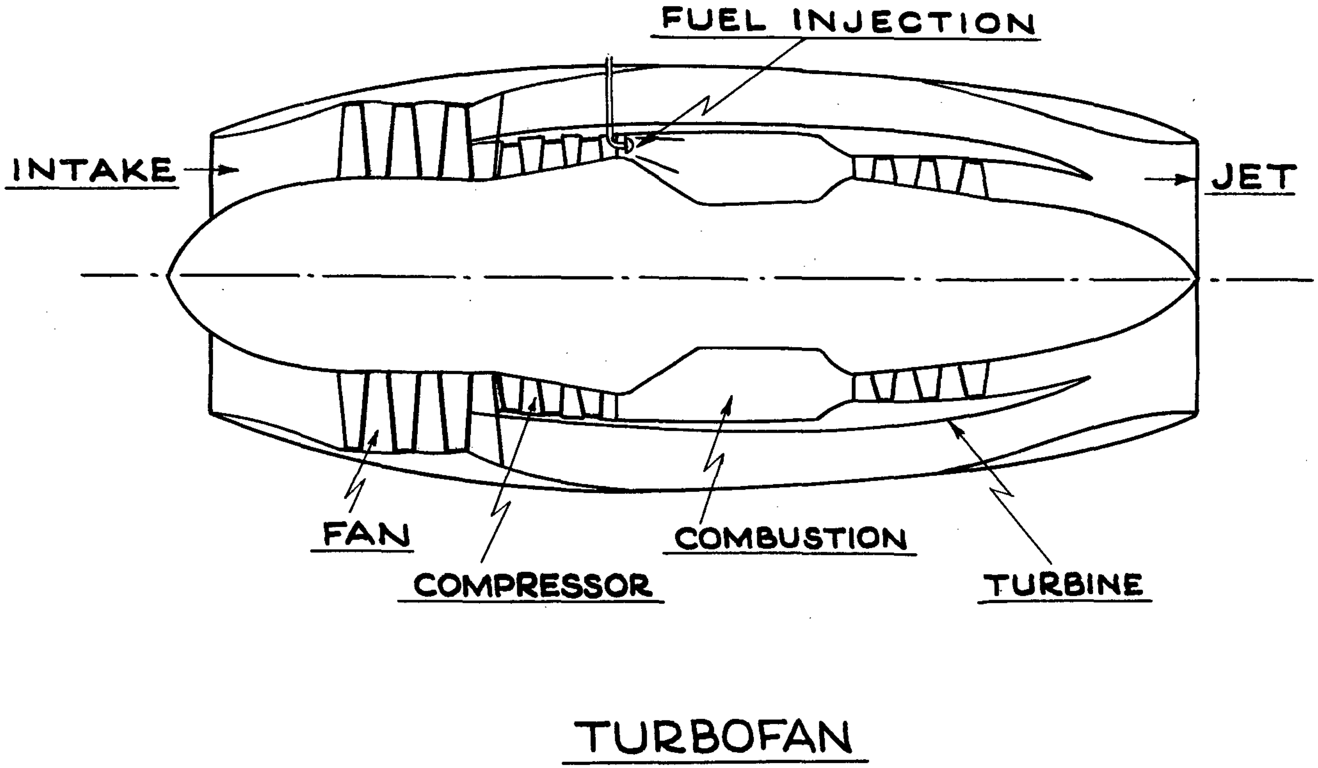
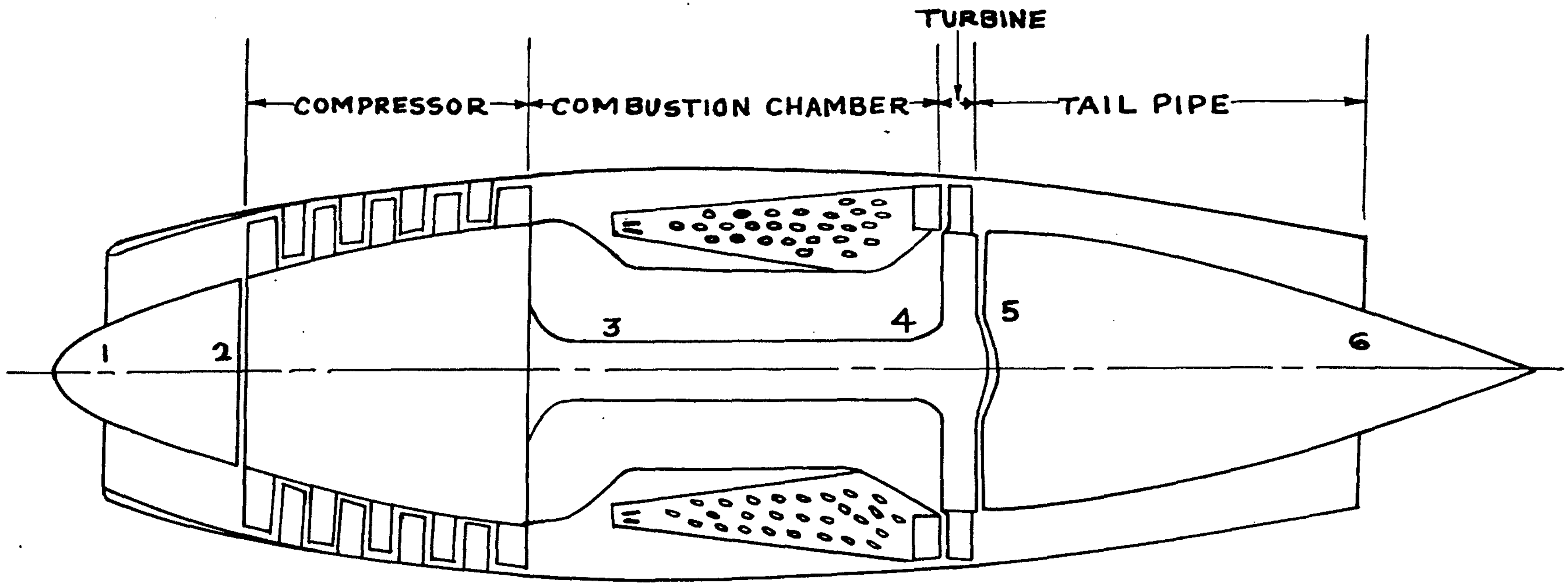


Figure 9 — Turbofan



TURBOJET

Figure 10 — Turbojet

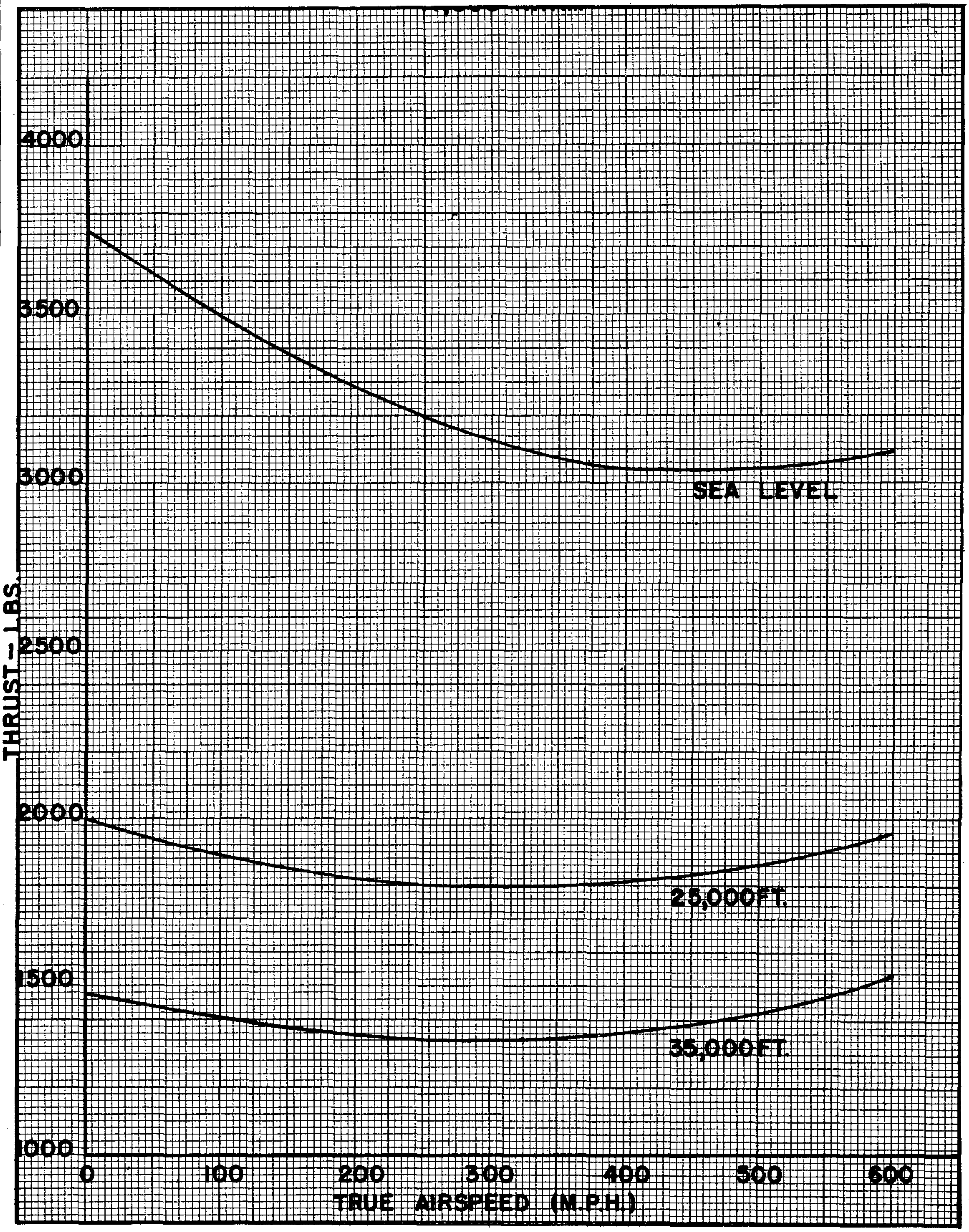


Figure 11 — Type I-40 Turbojet (11,500 RPM)

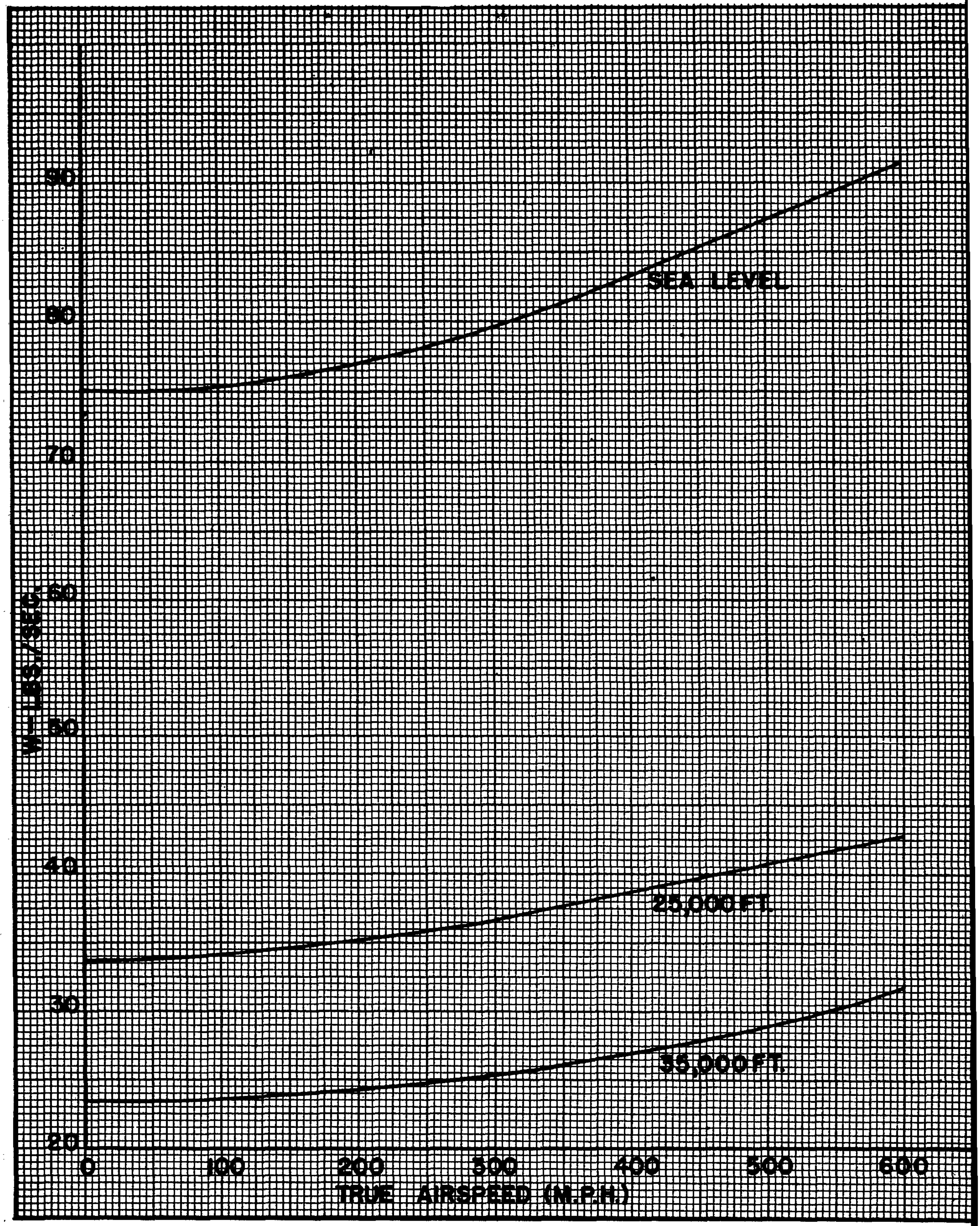


Figure 12 — Type I-40 Turbojet (11,500 RPM)

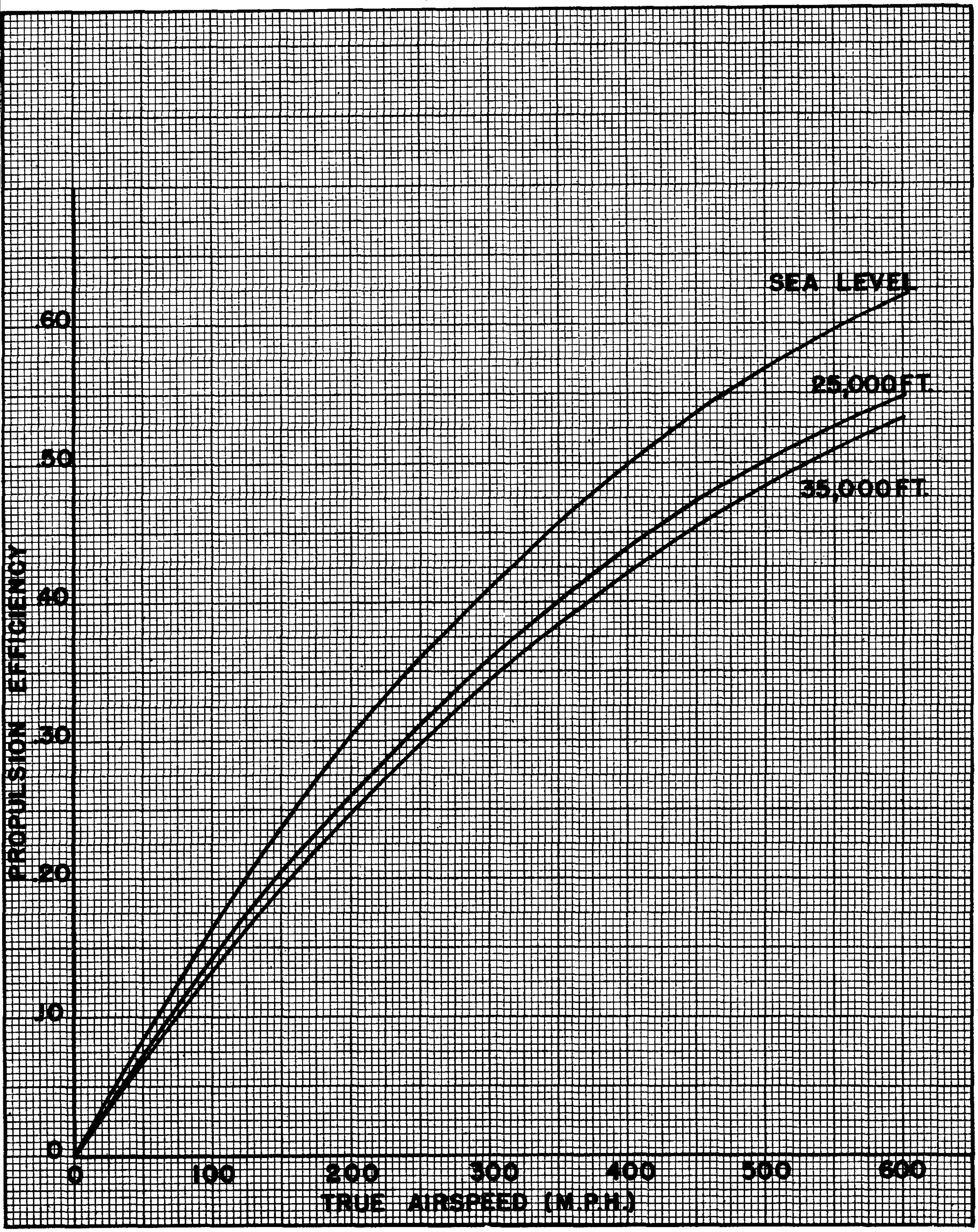


Figure 13 — Type I-40 Turbojet (11,500 RPM)

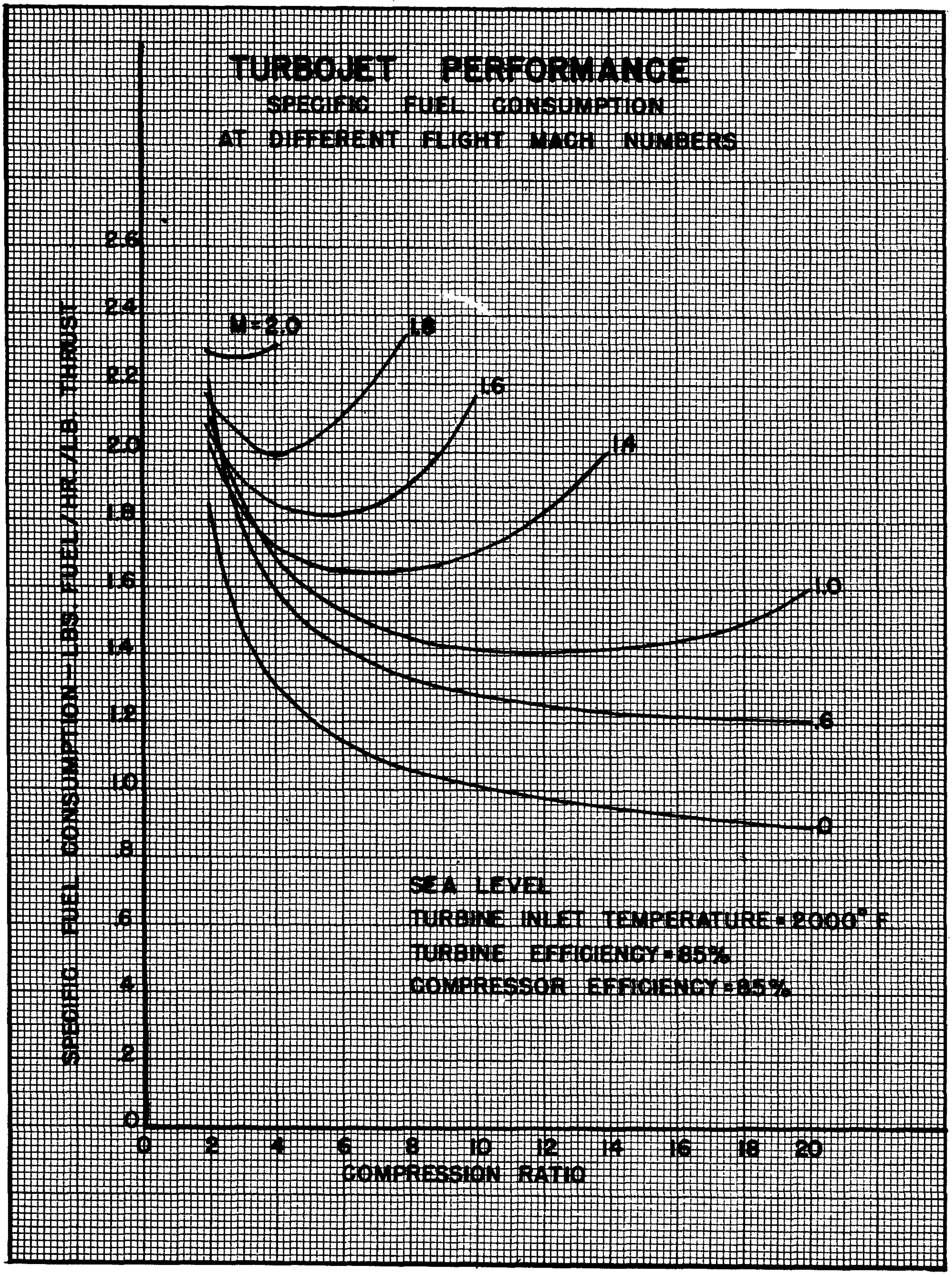


Figure 14 — Turbojet Performance (Specific Fuel Consumption at Different Flight Mach Numbers)

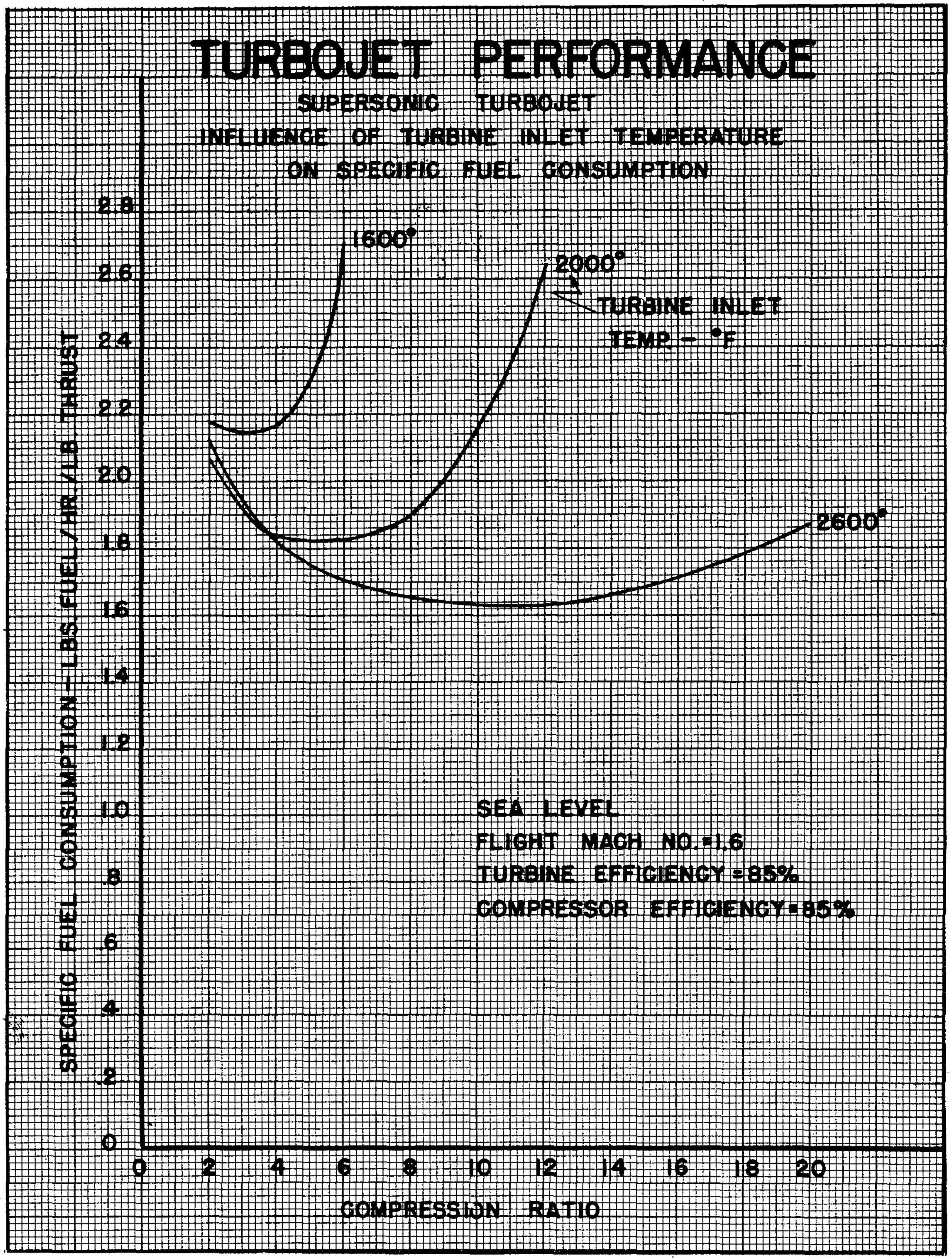


Figure 15 — Turbojet Performance (Supersonic Turbojet Influence of Turbine-Inlet Temperature on Specific Fuel Consumption)

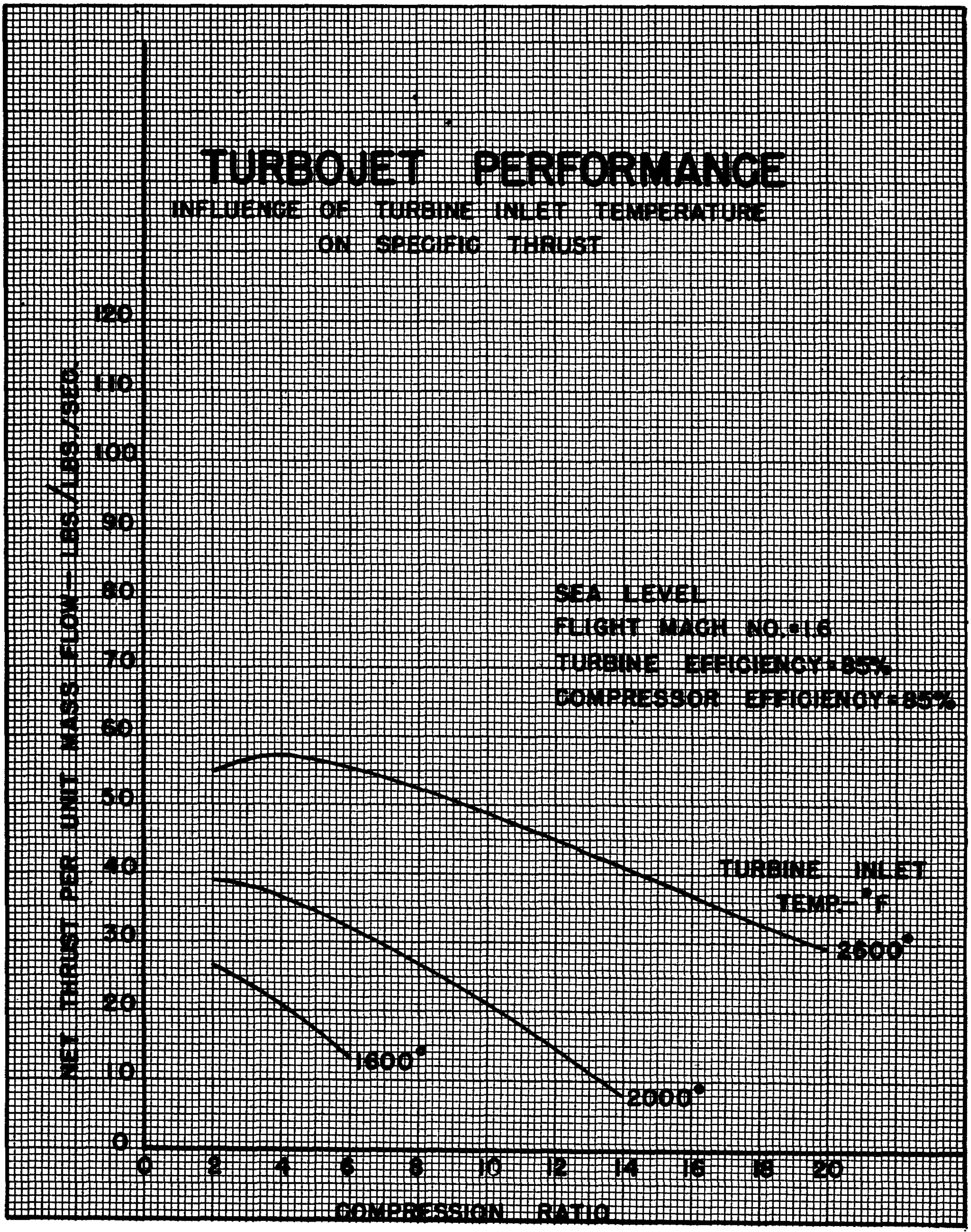


Figure 16 — Turbojet Performance (Influence of Turbine-Inlet Termperature on Specific Thrust)

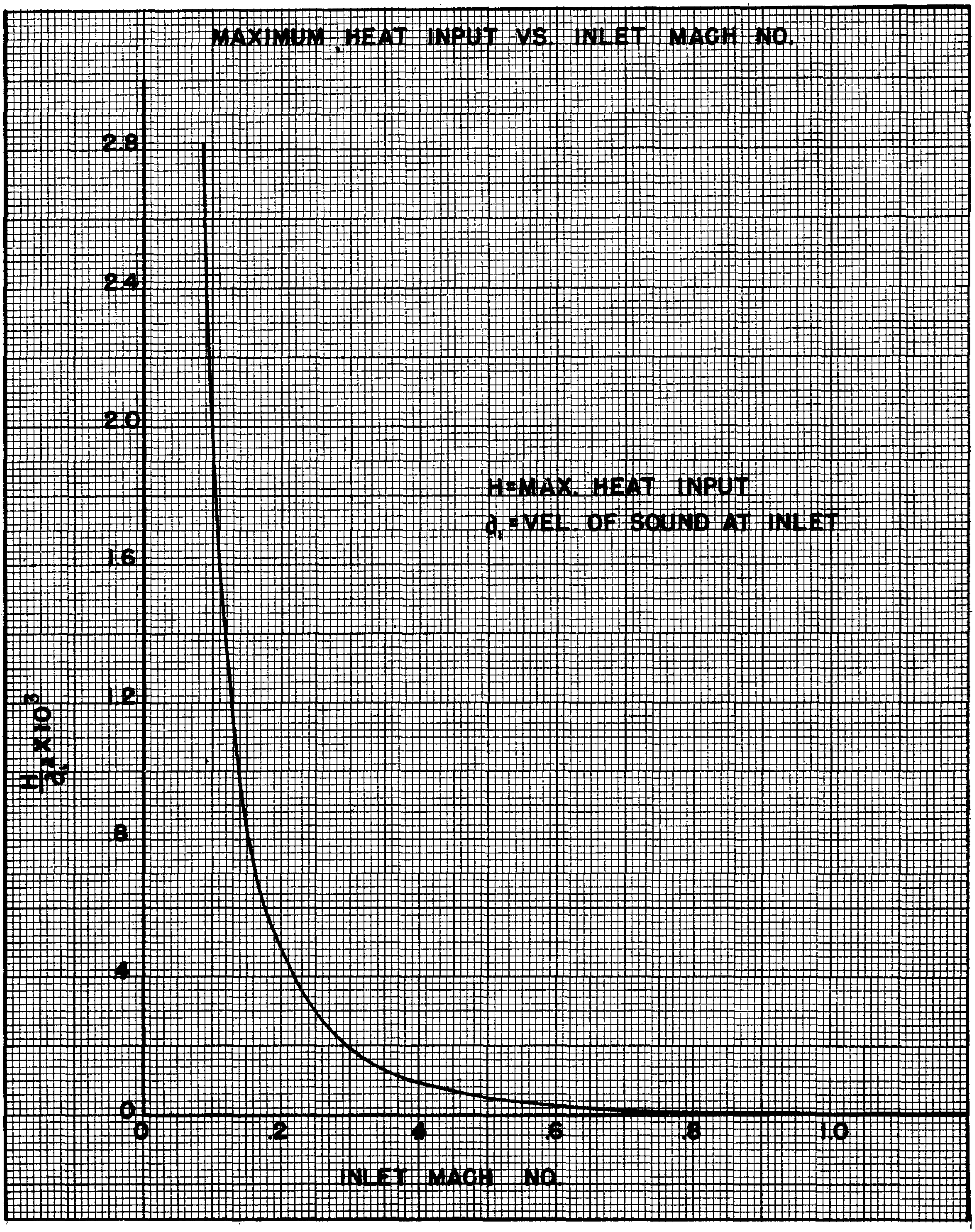
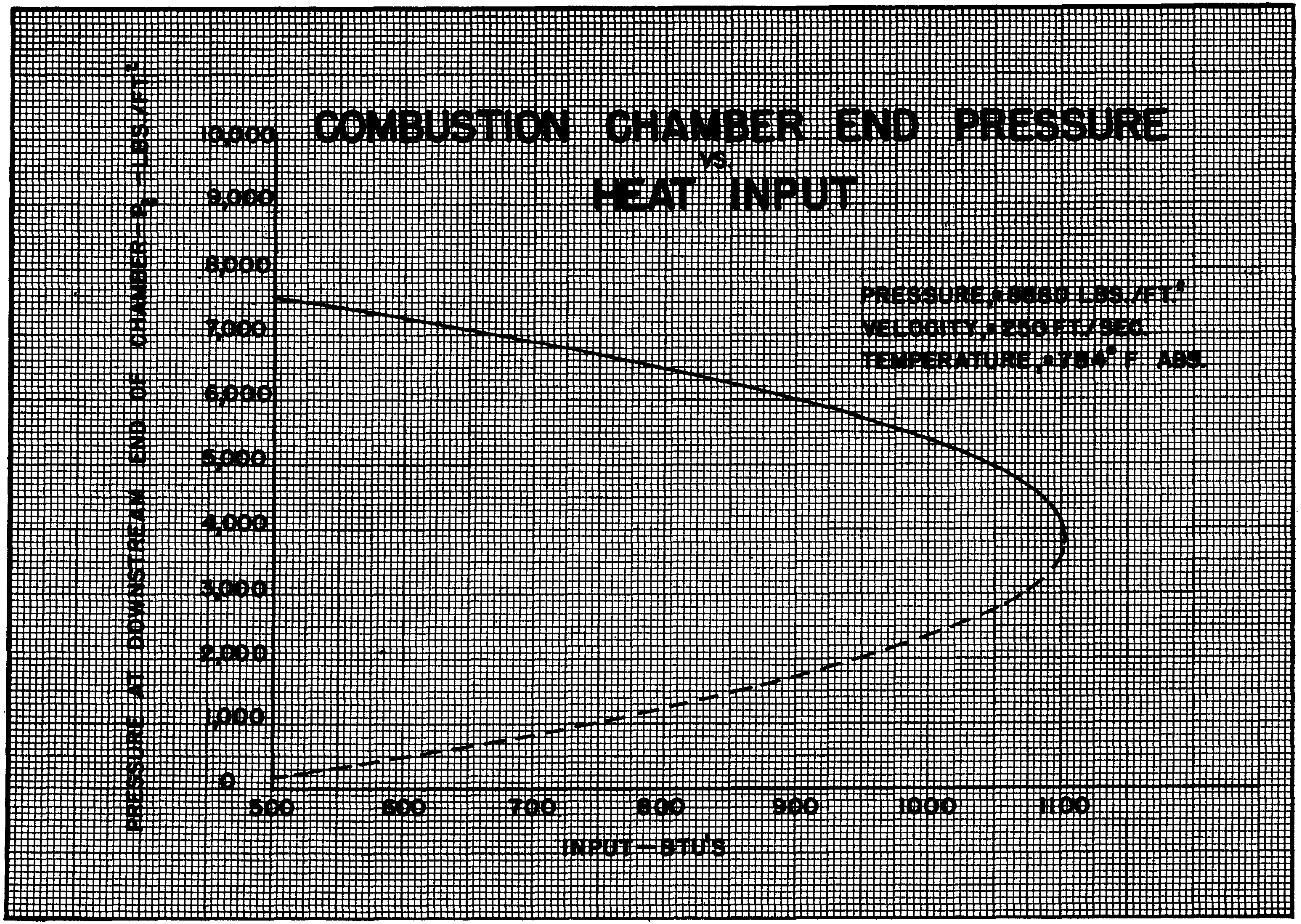


Figure 17 — Combustion Chamber Characteristics (Maximum Heat Input vs. Inlet Mach Number)

Discharge Mach no $M_2 = 1.0$



COMBUSTION CHAMBER END PRESSURE
VS.
HEAT INPUT
PRESSURE, = 5880 LBS./FT.²
VELOCITY, = 250 FT./SEC.
TEMPERATURE, = 784° F ABS.
PRESSURE AT DOWNSTREAM END OF CHAMBER—P_1—LBS./FT.²
10,000
9,000
8,000
7,000
6,000
5,000
4,000
3,000
2,000
1,000
0
500
600
700
800
900
1000
1100
INPUT—BTU'S

PART II

EXPERIMENTAL AND THEORETICAL PERFORMANCE OF AEROPULSE ENGINES

By

HSUE-SHEN TSIEN

PART II

EXPERIMENTAL AND THEORETICAL PERFORMANCE OF AEROPULSE ENGINES

DECEMBER 1945

THE PRESENT STATUS OF AEROPULSE

The German aeropulse and the American copy of it for the flying bomb are the first successful realizations of this type of power plant. The general dimensions are given in Fig. 1. The air is sucked into the combustion chamber by the vacuum created by exhaust of the previous cycle. The intake air passes the venturi where gasoline is continuously injected. The explosion of the air-fuel mixture raises the pressure in the combustion chamber to a high level and closes the spring valve at the intake. The gas is thus forced to expand through the exhaust duct and is discharged at high speed. This gives the propelling impulse. At the end of discharge, the inertia of the gas creates a vacuum in the combustion chamber and the engine is ready to start a new cycle again. The pressure in the combustion chamber is controlled by the rate of fuel injection, and this, in turn, controls the discharge velocity of the gas, and thus the propulsive thrust. To start the engine, a carefully adjusted amount of fuel is sprayed into the cold combustion chamber, so as to create a mixture of correct ratio around the spark plug. The spark plug ignites the mixture and the resulting strong explosion starts the cycle. The flow in the combustion chamber and the discharge duct is thus a pulsating one with very large amplitude, as shown by Figs. 2 and 3.

1. THE SPECIFIC FUEL CONSUMPTION.

The test results* on the German aeropulse are shown in Fig. 4, where the specific consumption in lb/hr/lb of gross thrust is plotted against the flight Mach number of the aircraft propelled by the power plant. The gross thrust is the total thrust of the unit without deducting the drag of the duct. At a given Mach number there are a number of points, each of which represents a different fuel injection pressure or fuel flow rate. As the fuel rate is increased, the pressure at the end of combustion is higher, and this tends to increase the efficiency of conversion of the heat energy to kinetic energy.

* "Preliminary Testing of German Robot Bomb Engines," Memorandum Report No. TSEPL-5-673-55, AAF (Nov 1944); "Twenty-Foot Wind-Tunnel Tests on the Jet Unit of a German Robot Bomb," Memorandum Report ENG.-51-6731-11, AAF (Aug 1944).

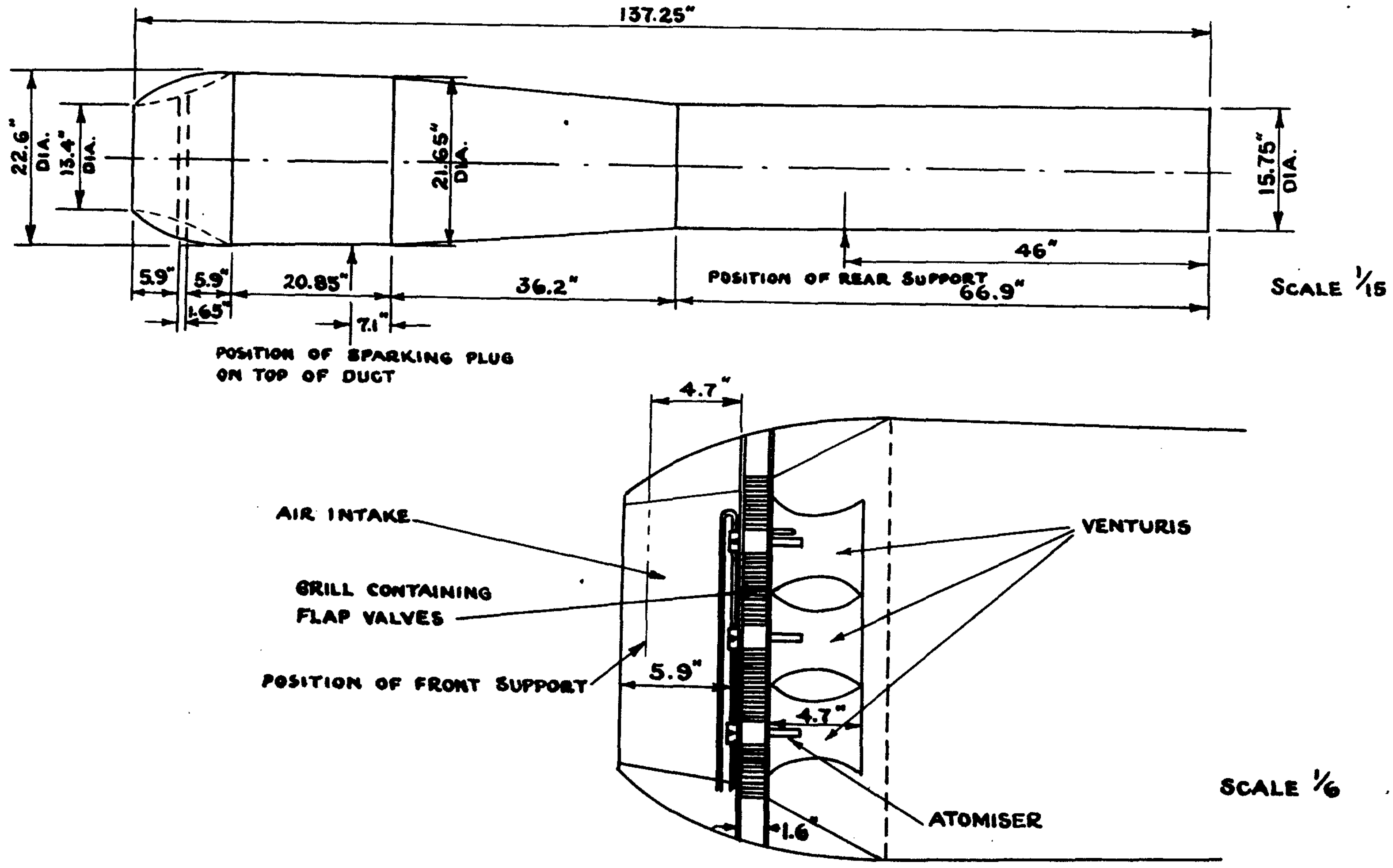


Figure 1 — German Aeropulse — General Arrangement of Duct

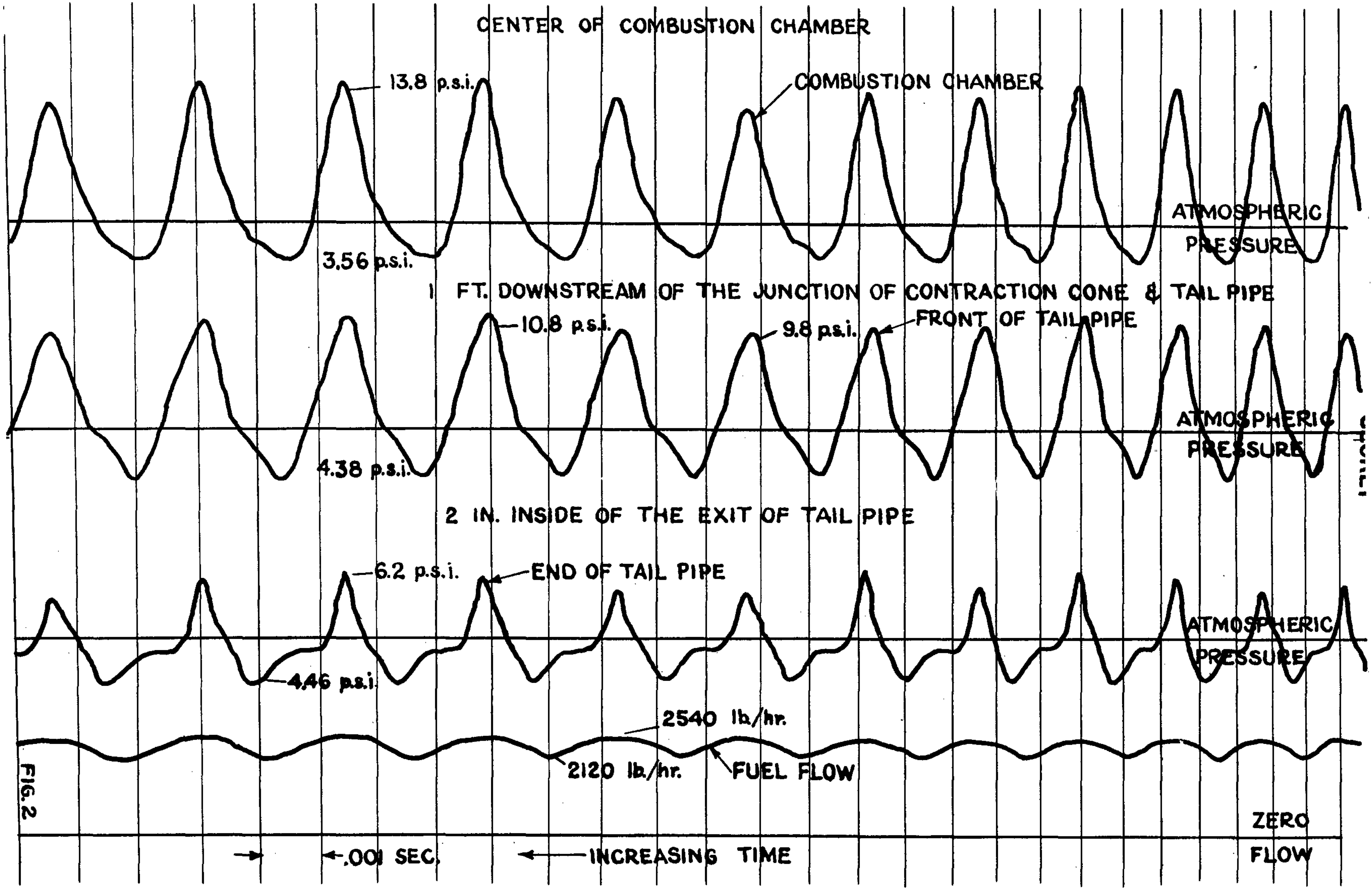


Figure 2 — 10 in. H_2O Ram and 23 p.s.i. Injection Nozzle Pressure. Pressures are Wall Statics

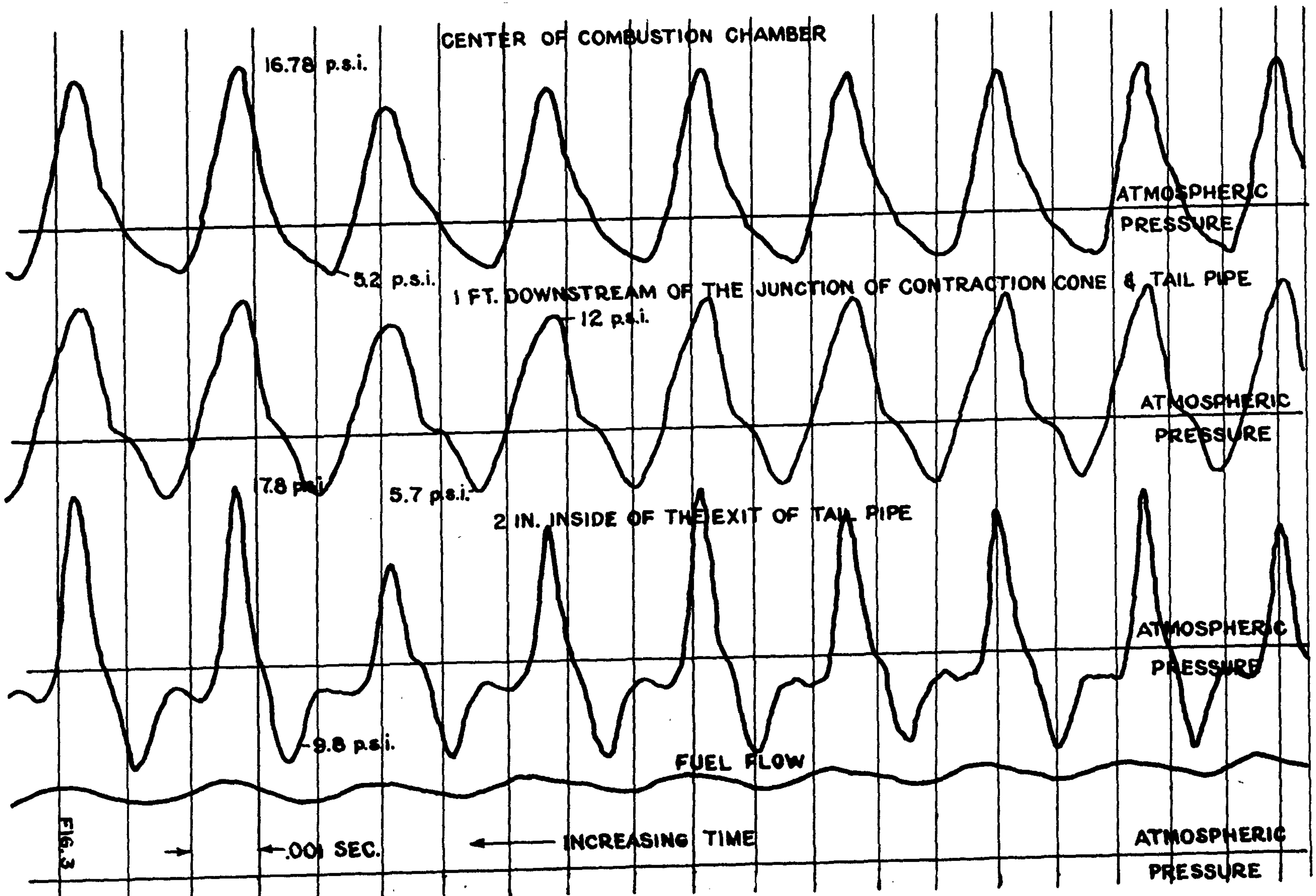


Figure 3 — 31 in. H_2O Ram and 37 psi Injection Nozzle Pressure. Pressures are Wall Statics

Therefore, it might be expected that the specific consumption will decrease with increase in fuel rate. In other words, the thrust output of the unit increases faster than the increase in fuel flow. This is actually borne out by the experiments, as the lower points in Fig. 4, at a given Mach number, correspond to higher fuel rates. It is seen that as flight Mach number or flight speed is increased, the specific consumption gradually decreases. The trend is represented by the heavy curve in the figure. At stationary conditions, the consumption is 5 lb/hr/lb-thrust. It decreases to 3.7 lb/hr/lb-thrust at a flight Mach number of 0.6, or 450 mph.

To extrapolate the specific consumption curve to higher flight numbers, it is necessary to have the guidance of theoretical calculation. Since the flow is nonsteady, the complete anlysis should include the inertia effects, and the calculation will be very complicated. To simplify the calculation the following assumptions will be made:

(a) The pressure in the combustion chamber at the end of the charging process or at the beginning of combustion is one-half of the stagnation pressure of the air stream, assuming isentropic compression.

(b) The combustion is carried out at constant volume and the combustion efficiency is 95 percent.

(c) The discharge process is quasisteady. In other words, the flow at each instant is assumed to be the same as that of a steady discharge through a Laval nozzle of perfect design, with the chamber pressure prevailing at that instant.

(d) The specific heats of the air and combustion products are assumed to be constants, but different values are used for air and for the combustion products.

(e) The mass flow of fuel is neglected with respect to the mass flow of air.

The basis for these assumptions will now be discussed.

The pressure in the combustion chamber at the end of the charging process depends upon two factors: the diffusor efficiency of the duct, and the throttling effect of the venturi with fuel jets. Since the duct is very short, most of the compression must occur outside of the duct with diverging streamlines. Therefore, aside from the possible effect of shock wave at supersonic flight velocities, the pressure in front of the spring valve should be very close to the stagnation pressure with isentropic compression. On the other hand, the throttling effect of the spring valve and venturi is very strong. Since the charging process is very fast, the flow velocity through the valve and venturi must approach that of sound. The pressure at the valve and venturi must be very close to one-half of the pressure in front of the spring valve. Due to the poor aerodynamic shape of the valves and the venturi, necessitated by the requirements for mixing of fuel and air, the pressure recovery must be inefficient. A rough approximation would be to assume no recovery. Thus the pressure in the combustion chamber at the end of the charging process is one-half of the stagnation pressure.

Since the combustion is explosive and occurs in very short intervals of time, the expansion of the product during combustion is negligible. Therefore, the assumption of constant volume should be a satisfactory approximation. The actual combustion process is quite complicated: During the expansion and dicsharge process, the gasoline injected into the combustion chamber is vaporized by the walls and the hot gas remaining in the chamber. The result is an over-rich mixture, which is pre-

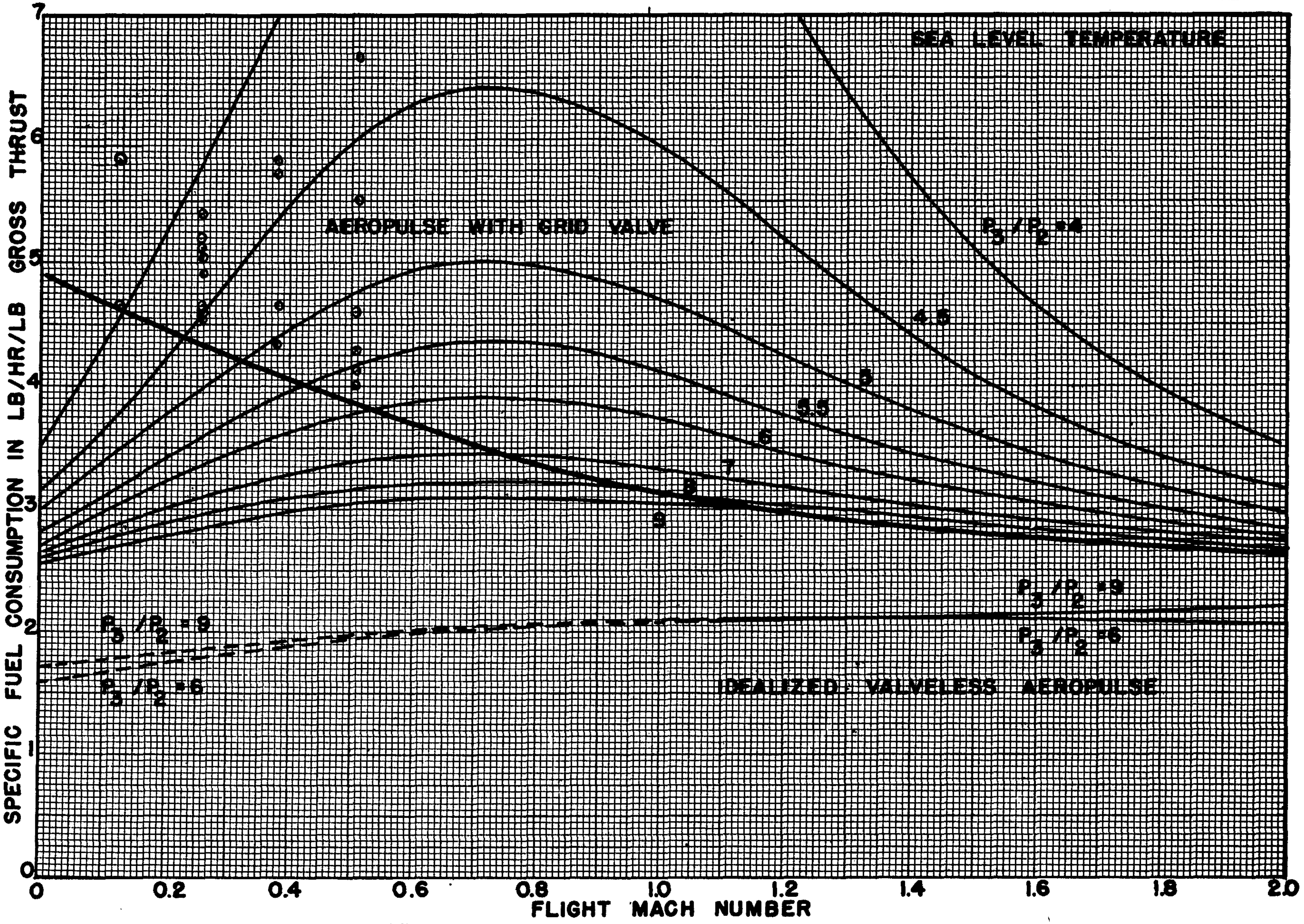


Figure 4 — Test results on the German Aeropulse

sumably partially oxidized, i.e., the large hydrocarbon molecules are partially broken down to more active elements. When the discharge process is completed, the low pressure then prevailing in the chamber will open the spring valve, and fresh air in strong eddies created by the venturi will be admitted to the chamber. The mixing of the fresh air with the partially oxidized and activated hydrocarbon vapor immediately starts the rapid combustion with a rise in pressure. Of course, if detonation occurs, much higher local pressures can be attained than by smooth combustion, as assumed here. However, actual measurements do not indicate the occurrence of detonation. Thus the assumption of constant volume combustion should be satisfactory.

The assumed discharge process really corresponds to that of a slow discharge through a small nozzle fitted to a large chamber. In the slow discharge, the inertia effect is small and can be neglected. Then the quasisteady process is an exact representation of the physical phenomenon. The assumption adopted is thus to approximate the impulse of a rapid discharge process by the impulse of a slow discharge process. This approximation should be satisfactory, as generally the impulse of a jet propulsion device is little influenced by the rate of the process.

The last two assumptions of constant specific heats and constant mass flow are generally adopted and are believed to have little influence on the results.

The results are shown in Fig. 4. The curves correspond to different values of the pressure ratio after combustion and before combustion. For a given pressure ratio, the specific consumption first increases with the flight speed, but then reaches a maximum value and decreases with further increase in flight speed. The larger pressure ratios give lower fuel consumption than expected. However, in order to explain the experimental data, it seems that the maximum pressure ratio that can actually be obtained in the German aeropulse is a function of the forward speed. At lower speed, this ratio is smaller than four. This is in accordance with the pressure record shown in Figs. 2 and 3. However, as the speed increases, the combustion can be pushed to higher limits and the pressure ratio is increased, giving a decrease in the actual fuel consumption. On the other hand, with gasoline as the fuel, the maximum combustion temperature is roughly 5000°F. In other words, the pressure ratio cannot increase indefinitely, but must be limited to a value lower than nine. Hence, the theoretical specific consumption corresponding to the pressure ratio nine forms the lower limit of the consumption of the German aeropulse. This is the basis for the heavy curve in Fig. 4, representing the probable trend of consumption with increase in flight velocity. Approximately then, the specific consumption of the German aeropulse is not likely to be lower than 3 lb/hr/lb-thrust at supersonic speeds.

2. FREQUENCY OF PULSATION AND THRUST

The simple theory of aeropulse outlined in the preceding section, being based upon a quasisteady discharge process, fails completely in predicting the frequency which is directly associated with the nonsteady pulsating flow. In order to have an estimate of the frequency, another very simple picture of operation will be adopted. This is the assumption of very small pressure ratio, i.e., very small pressure amplitude. Then the frequency can be calculated by the elementary considerations applicable to small pressure amplitudes, i.e., the sound vibration in a pipe. Since experiments (Ref.

cited on page 41) show that the frequency of the aeropulse varies very little with changes in the fuel rate or the pressure ratio; the frequency so calculated for very small pressure amplitudes should be representative for that of large pressure amplitudes.

Consider the aeropulse as an organ pipe closed at the end where the spring valve is located and open at the other end. Then the pulsation in the pipe can be considered as a quarter wavelength oscillation with a maximum pressure amplitude at the closed end and a zero pressure amplitude, but maximum velocity amplitude at the open end, as shown by Fig. 5. If a* is the velocity of sound propagation and L the length of the pipe, the frequency f is

$$f = \frac{a^*}{4L} \text{ cps.} \tag{1}$$

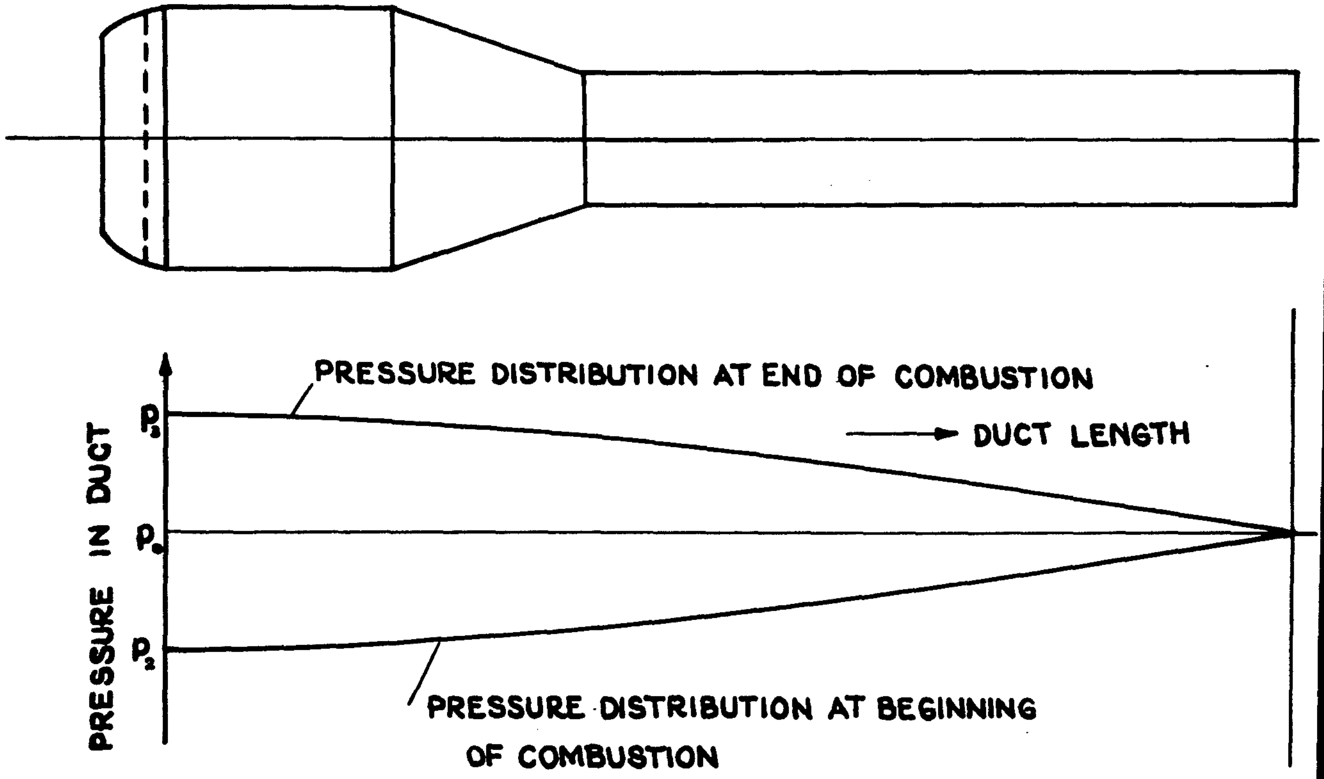


Figure 5

For the application to the aeropulse, a* should correspond to the mean conditions of the flow in the duct. It is estimated that the German aeropulse with no forward velocity should have a frequency of f = 50.1 cps. This checks very well with the measured value of 46 cps. Most of the difference is perhaps accounted for by the combustion period, which, although short, is not negligible. At higher fuel rates, the combustion period is lengthened. This tends to lower the frequency. Experiments

show that the frequency at high fuel rates is actually slightly lower, in spite of the higher sound velocity at higher temperatures, due to the richer mixture.

The average thrust of the aeropulse depends on the rate of airflow and the mixture ratio. The rate per second at which the air is taken into the duct is, in turn, dependent upon the charge per cycle and the number of cycles per second. However, all three factors are interrelated. For instance, if the number of cycles per second is very large. the charge pressure in the chamber will be low due to the necessary rapid acceleration to push the air into the chamber. In other words, the charge per cycle tends to decrease as the frequency is increased. Furthermore, the combustion process also tends to limit the mixture ratio that could be effectively used to lower values if the frequency were increased as the combustion time grows shorter. Thus the average thrust of the aeropulse, being an increasing function of the product of all three factors, has a maximum with respect to the rate of fuel injection and with respect to the frequency, which depends on the length of the tail pipe. The prediction of the thrust of an aeropulse is thus rather complicated. The situation is very similar to the case of the reciprocating gasoline engine. The calculation of horsepower output of the reciprocating engine is generally based upon empirical data, as the effects of volumetric efficiency or breathing capacity and combustion conditions are very difficult to calculate.

The experimental data available at present on the thrust output of the German aeropulse is plotted in Fig. 6. The gross thrust is presented in two forms, C_F and K_F. C_F is the ratio of the maximum gross thrust to the product of the dynamic pressure and the sectional area of the combustion chamber or the frontal area. K_F is the ratio of the maximum gross thrust to the product of the atmospheric pressure and the sectional area of the combustion chamber. Thus

$$C_F = \frac{\text{gross thrust}}{1/2 \text{ air density x (velocity)}^2 \text{ x frontal area}} \tag{2}$$

$$K_F = \frac{\text{gross threat}}{\text{atmospheric pressure x frontal area}} \tag{3}$$

Both quantities are plotted against the flight Mach number and are related by the formula,

$$C_F = K_F \frac{2}{\delta M_o^2} \tag{4}$$

where δ is the ratio of specific heats for air and M_o is the flight Mach number. Due to the fact that the aeropulse has a static thrust, the value of K_F is not zero at zero velocity. Therefore C_F become infinite when $M_o = O$.

As stated previously, the prediction of the thrust of aeropulse is very difficult. Therefore, to estimate the thrust coefficients at higher flight velocities, two assumptions have to be made:

(a) The frequency of the unit remains the same as at low speed.

(b) The maximum thrust operating conditions follow the heavy line in Fig. 4.

The first assumption agrees with the observational data available and leads to the conclusion that the air mass flow through the aeropulse is proportional to the density of the air in the combustion chamber at the end of the charging process. The second assumption seems justified by the test data, as explained previously.

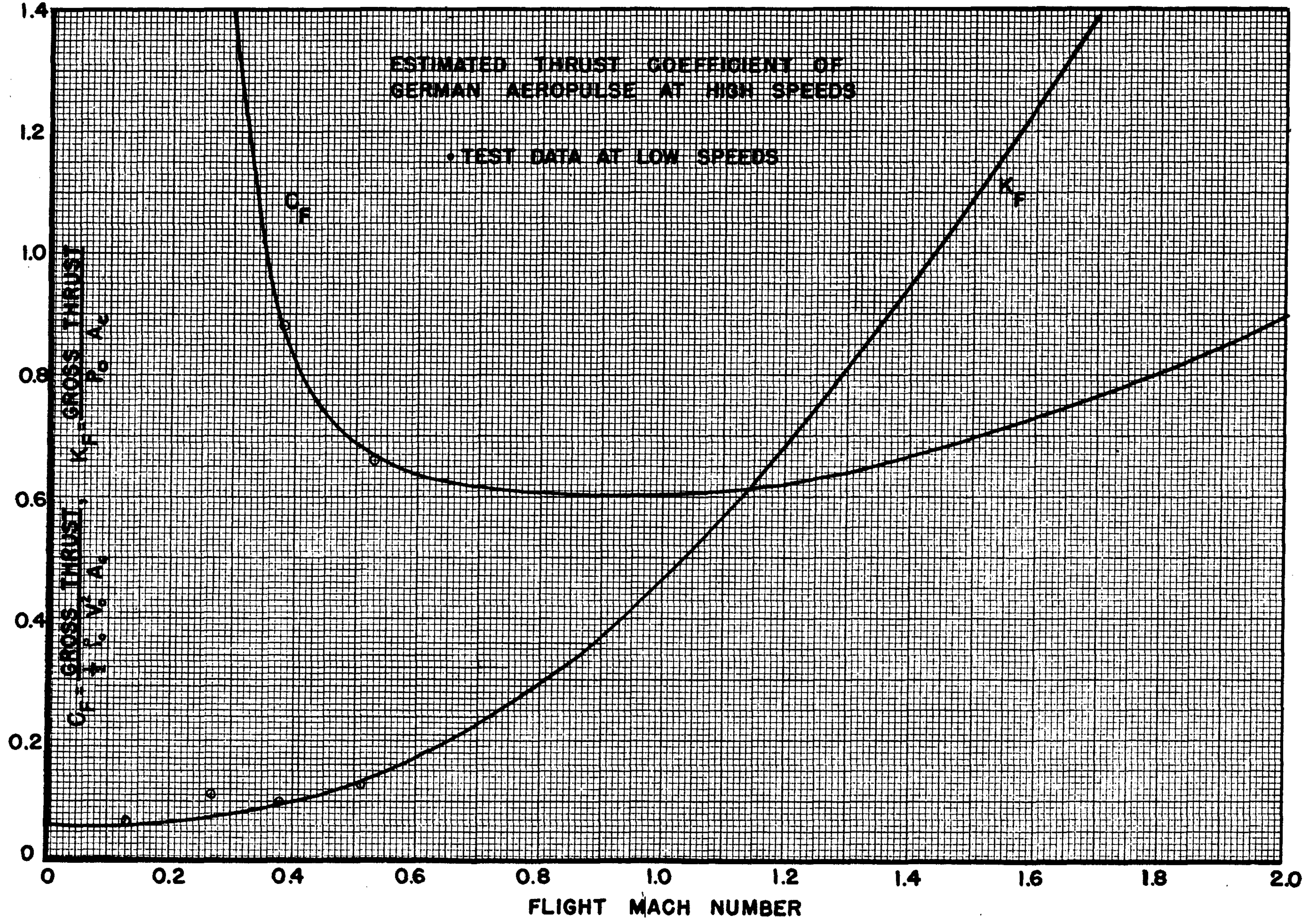


Figure 6 — Estimated Thrust Coefficient of German Aeropulse at High Speeds

With these assumptions, the thrust coefficient curves can be extrapolated as shown in Fig. 6. It is seen that for the German aeropulse at supersonic speed, the value of the thrust coefficient C_F is almost constant and remains at about 0.70.

POSSIBLE IMPROVEMENTS OF THE EXISTING FORM OF AEROPULSE

To increase the thrust and to reduce the fuel consumption of the present-day aeropulse as described in the previous paragraphs, either the air flow or the combustion pressure should be increased and the external drag of the duct reduced. To increase the air flow, the effective flow area into the combustion chamber must be enlarged. The resultant reduction in the throttling action of the grid valve will also raise the pressure in the combustion chamber at the end of the charging process and hence give higher combustion pressure and better fuel economy. For instance, by removing part of the rib* in the grid of the valve, the static thrust can be increased from 660 to 880 lb. The specific consumption can be similarly reduced from 3 lb/hr/lb of gross thrust to 2.8 lb/hr/lb gross thrust. To reduce the external drag, the unit can be mounted internally in the fuselage. But this is not possible with the present construction material, due to the overheating caused by the lack of cooling air flow.

If air alone could be introduced into the duct behind the explosive air-fuel mixture, then when the mixture is burned, it would act as a piston to push out the air column. The total air mass per cycle is thus increased, with resultant larger momentum and better efficiency. The air and air-fuel mixture must be separated for two reasons: (1) A very lean mixture will not burn properly. (2) Even if the mixture burns properly, the explosion pressure would be too low for effective energy utilization. This form of augmentation really already occurs in the present aeropulse to a certain degree, as during the charging time some atmospheric air flows into the duct through the rear opening. This part of the charge contains no gasoline (which is only injected into the combustion chamber) and serves as the air for augmentation. For air to enter the duct through the rear opening, the direction of the flow has to be reversed, and this becomes more difficult as the flight speed is increased. Therefore, this natural augmentation effect decreases at higher velocities. To remedy this, it has been suggested that two ducts be used, mounted one in front of the other, as shown in Fig. 7. Then during the suction or charging period, the air flows into the second grid B and fills the second duct with augmentation charge. However, this idea has yet to be exhaustively tested.

The combustion pressure can also be raised by using a fuel which tends to detonate. Of course, for such fuels the injection has to be timed properly instead of using the present continuous uncontrolled flow. This may lead to the further necessity of using a timed spark plug to ignite the mixture. Whether there would be a large enough increase in performance by this method to justify the complications, only experiments can determine. The weight of the present aeropulse is about 300 lb, for a thrust of

* This was done by Eisla and G. Dietrich of DFS at Ainring, Germany, during 1944.

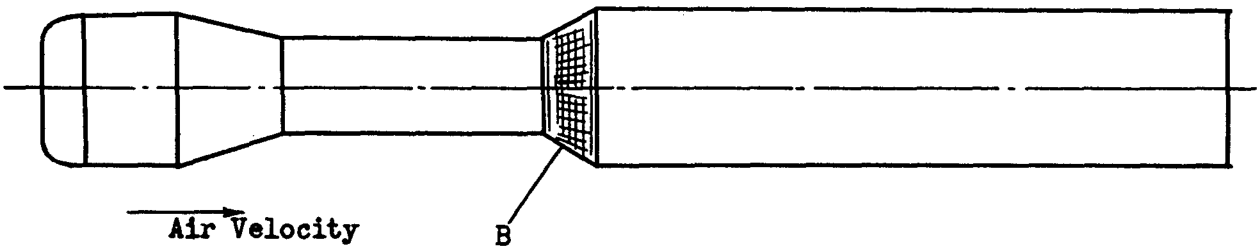


Figure 7

70 lb. With better materials and method of construction, this value could be reduced. This is another possible direction of development.

VALVELESS AEROPULSE

An obvious improvement for the present aeropulse is the complete elimination of the spring valve and the venturi. If the valve and venturi can be discarded, then the strong throttling effect can be avoided. The result would be an increase in the chamber pressure at the end of the charging process, to the stagnation pressure, or twice the present value. This would give a higher density of air in the chamber, and hence larger mass flow and higher thrust. Furthermore, the pressure at the end of combustion for a given temperature rise will also be increased and thus more efficient expansion and lower specific consumption will be attained. Of course, at low flight speeds a valveless aeropulse cannot be very efficient, due to the partial expansion of the combustion products through the front opening and the loss of impulse or thrust of the unit. At supersonic flight velocities, such losses can be eliminated, as the explosion pressure travels with a definite speed only slightly higher than that of sound. If high flow velocity can be maintained in the duct from front to rear, the explosion pressure can never travel to the front part of the duct. Therefore, the expansion of the hot gas will be directed only toward the rear. In other words, the inertia of the high-speed air flow in the front part of the duct acts as the valve, without the necessity of a mechanical valve. In fact, with proper design of the duct, this possibility exists even at high enough subsonic speeds.*

1. IMPROVEMENT IN SPECIFIC FUEL CONSUMPTION

By using the same simplifying assumptions as before, except that the pressure in the combustion chamber at the end of charging process is now equal to the stagnation pressure, the specific fuel consumption can be easily calculated. The result is shown in Fig. 4. It is shown that here an increase in the combustion pressure ratio, i.e., the ratio of pressure before combustion and after combustion, tends to increase

* "Note on Valveless Aero- and Hydro-Pulse Motors," AMP Memo 137.1M, AMG-NYU No. 120, Applied Mathematics Group, New York University, May 1945.

the specific fuel consumption. However, the difference for a rather large change from pressure ratio nine to pressure ratio six is quite small. The parts of the curves for flight Mach number less than unity are drawn in broken lines, to indicate that the simple theory does not correspond to a real situation, due to the neglect of expansion and escape of the combustion products from the front opening of the duct. Thus, in general, the idealized valveless aeropulse will give a specific consumption equal to 2 lb/hr/lb of gross thrust in the supersonic region. Actually, of course, minor losses are inevitable and a valveless aeropulse will have a specific consumption somewhat higher than this, i.e., between 2 and 3 lb/hr/lb-thrust. The higher value is the specific consumption of the valved type, for example the German aeropulse for the flying bomb.

2. INCREASE IN THRUST COEFFICIENT

For the valveless aeropulse the throttling effect of the spring valve and the venturi is eliminated and the pressure of air in the combustion chamber at the beginning of combustion is twice as high as for the valved type. Since the temperature of the air is the same as in the previous case, the density of air in the chamber is also twice as large as for the valved type. Thus the mass of air flowing through a duct of given size will be doubled. If the heat added per unit mass of air is made the same by using the same combustion pressure ratio, the total heat added will be twice as large. On the other hand, the specific consumption is lowered by a factor of two-thirds, as stated in the preceding paragraph. Therefore, the thurst produced for a given size of the duct will be three times as large as the valved type. Actually, if losses occur, the difference will not be as large. However, it seems that a C_F as high as 1.0 for a valveless aeropulse in supersonic flight can be considered reasonable.

CONCLUDING REMARKS

In the foregoing sections, the possibilities of further improving the aeropulse are discussed. It seems that with these developments a lightweight aeropulse could be made, with a weight as low as 0.16 lb/lb gross thrust at sea level, especially at high speeds. The specific consumption based upon the gross thrust could be lowered to about 2 lb/hr/lb-thrust. However, to carry out these investigations, a supersonic wind tunnel of large enough test section so that a complete aeropulse can be tested with burning, is a necessity. The strongly pulsating flow around and inside an aeropulse indicates the close relation between the inside flow through the duct and the outside flow around the duct. Therefore, only a complete model test with the model submerged in an air stream can reproduce the correct flow conditions. Static tests, or tests with air flow through the engine, would be misleading and unreliable.

APPENDIX I

A Simple Theory for the Aeropulse

Let the subscript o denote quantities corresponding to the free atmospheric conditions, the subscript 1 denote the stagnation conditions when the spring valves are closed, the subscript 2 denote the conditions in the combustion chamber at the end of the charging process, and the subscript 3 denote the conditions at the end of combustion. Since the compression from free stream to the stagnation pressure is assumed to be isentropic,

$$\frac{p_1}{p_o} = \left(1 + \frac{\delta - 1}{2} M_o^2\right)^{\frac{\delta}{\delta - 1}} \tag{5}$$

and the temperature ratio is

$$\frac{T_1}{T_o} = 1 + \frac{\delta - 1}{2} M_o^2 \tag{6}$$

According to the assumption stated in the main text,

$$p_2 = \frac{1}{2} p_1 \tag{7}$$

The temperature T_2, being a representation of the total energy of the gas at rest, must be the same as T_1, since no appreciable heat loss can occur.

$$T_2 = T_1 \tag{8}$$

Since the combustion is assumed to be carried out at constant volume, the heat added h per unit mass

$$h = C'_v \left(T_3 - T_2\right) = \frac{1}{\delta'} C'_p T_3 \left(1 - \frac{T_2}{T_3}\right). \tag{9}$$

The primed quantities refer to the combustion products. Since the combustion is carried out at constant volume

$$\frac{T_2}{T_3} = \frac{p_2}{p_3}. \tag{10}$$

Therefore,

$$h = \frac{1}{\delta'} C'_p T_3 \left(1 - \frac{p_3}{p_2}\right). \tag{11}$$

To calculate the discharge process, the quasisteady flow is assumed. Thus if v is the discharge velocity corresponding to p, then

$$v = \sqrt{\frac{2\delta'}{\delta' - 1} \frac{p}{\rho} \left[1 - \left(\frac{p_o}{p}\right)^{\frac{\delta' - 1}{\delta'}}\right]} \tag{12}$$

The impulse due to a discharge dm at this velocity is

$$dI = vdm \tag{13}$$

Let m be the mass before the removal of dm, the ratio of density in the chamber after the removal of dm and that before removal is then

$$\frac{m - dm}{m} \tag{14}$$

Similarly, the pressure ratio is

$$\frac{p + dp}{p} \tag{15}$$

Since the process in the combustion chamber is isentropic, we have

$$\frac{p + dp}{p} = \left(\frac{m - dm}{m}\right)^{\delta_1} \tag{16}$$

Thus

$$\delta' \frac{dm}{m} = -\frac{dp}{p} \tag{17}$$

Now $m = \rho V$, where V is the volume of the combustion chamber.

Therefore

$$d\,I = -\sqrt{\frac{2\delta'}{\delta'-1}\frac{p}{\rho}\left[1 - \left(\frac{p_o}{p}\right)^{\frac{\delta'-1}{\delta'}}\right]}\frac{1}{\delta'}\rho V\frac{dp}{p} \tag{18}$$

To find the total impulse due to the discharge, we have to integrate dI for pressure variations from the initial pressure p_3 to the final p_o. Therefore

$$I = \frac{1}{\delta'}\sqrt{\frac{2\delta'}{\delta'-1}}V\rho_3\sqrt{\frac{p_3}{\rho_3}}\int_{p_o/p_3}^{1}\sqrt{\frac{1}{\eta^{\frac{\delta'-1}{\delta'}}}\left[1 - \left(\frac{p_o}{p_3}\right)^{\frac{\delta'-1}{\delta'}}\frac{1}{\eta^{\frac{\delta'-1}{\delta'}}}\right]}\,d\eta \tag{19}$$

where $\eta = p/p_3$. But $V\rho_3$ is the total mass in the combustion chamber at the beginning of discharge, and $\left(\delta'\frac{p_3}{\rho_3}\right)^{1/2}$ is the velocity of sound, A_3, corresponding to the conditions in the combustion chamber at the end of combustion. Thus if v_e, denotes the "effective exit velocity," then

$$\frac{v_e}{a_3} = \frac{1}{\delta'}\sqrt{\frac{2}{\delta'-1}}\int_{p_o/p_3}^{1}\sqrt{\frac{1}{\eta^{\frac{\delta'-1}{\delta'}}}\left[1 - \left(\frac{p_o}{p_3}\right)^{\frac{\delta'-1}{\delta'}}\frac{1}{\eta^{\frac{\delta'-1}{\delta'}}}\right]}\,d\eta \tag{20}$$

For the case of $\delta = 4/3$, the integral is of the closed form

$$\frac{v_e}{a_3} = \frac{6\sqrt{6}}{7}\sqrt{1 - \left(\frac{p_o}{p_3}\right)^{\frac{1}{4}}}\left[1 - \frac{1}{5}\left(\frac{p_o}{p_3}\right)^{\frac{1}{4}} - \frac{4}{15}\left(\frac{p_o}{p_3}\right)^{\frac{2}{4}} - \frac{8}{15}\left(\frac{p_o}{p_3}\right)^{\frac{3}{4}}\right] \tag{21}$$

If the average mass flow is m, then the thrust is m $(v_e - v_o)$. Let H be the heat value of the fuel per lb and η_b the combustion efficiency, then

$$s = \frac{3600\eta}{778H\eta_b(v_e - v_o)} \tag{22}$$

After some reduction, this equation can be written as

$$s = \frac{3600}{778H\eta_b} \frac{a_o}{\delta'(\delta'-1)} \frac{\frac{a_3}{a_o}\left(1 - \frac{p_2}{p_3}\right)}{\frac{v_e}{a_3} - M_o\left(\frac{a_o}{a_3}\right)} \tag{23}$$

In this formula the ratio of sound velocity is given by

$$\left(\frac{a_3}{a_o}\right)^2 = \frac{C_p'}{C_p} \frac{\delta'-1}{\delta-1} \left(\frac{p_3}{p_2}\right) \left(\frac{T_1}{T_o}\right) \tag{24}$$

Fig. 4 is calculated on the following assumed values for the constants:

$$\begin{aligned} C_p' &= 0.276 \text{ BTU/lb/°F} \\ C_p &= 0.243 \text{ BTU/lb/°F} \\ \delta &= 1.405, \qquad \delta' = 4/3 \\ H &= 18{,}700 \text{ BTU/lb} \end{aligned} \tag{25}$$

and

$$\eta_b = 95\%.$$

APPENDIX II

Frequency of Aeropulse

For the application to aeropulse, a* must correspond to the mean conditions of the flow in the duct. Since the temperature in the duct at the end of expansion is

$$T_3 \left(\frac{p_o}{p_3}\right)^{\frac{\delta'-1}{\delta'}} \tag{26}$$

the mean value a* for the velocity of sound is

$$a* = \frac{1}{2} a_o \left(\frac{a_3}{a_o}\right) \left[1 + \left(\frac{p_o}{p_3}\right)^{\frac{\delta'-1}{2\delta'}}\right] \tag{27}$$

For instance, if $p_3/p_2 = 4$, $M_o = 0$, then $p_3/p_2 = 2$ (28)
with $\delta' = {}^4/_3$, $a* = 1.849\ a_o$. The German aeropulse has an effective length, L = 10.3 ft, and for standard sea level conditions, $a_o = 1/20$ ft/sec. Thus

$$f = \frac{1.849 \times 1120}{4 \times 10.3} = 50.1 \text{ cps.} \tag{29}$$

APPENDIX III

Thrust of an Aeropulse at High Flight Speeds

To extrapolate the thrust for higher flight velocities, it is assumed that the frequency of the unit will remain approximately constant. Then the volume flow of air will also be constant. However, the density of the air is proportional to

$$\left(1 + \frac{\delta - 1}{2} M_o^2\right)^{\frac{1}{\delta' - 1}} \tag{30}$$

Thus the air mass flow is proportional to the same factor. On the other hand, the heat added per unit mass of air is proportional to the temperature difference T_3-T_2, or proportional to

$$\left(1 + \frac{\delta - 1}{2} M_o^2\right)\left(\frac{p_3}{p_2} - 1\right) \tag{31}$$

Hence, the total heat added is proportional to

$$\left(1 + \frac{\delta - 1}{2} M_o^2\right)^{\frac{\delta'}{\delta' - 1}}\left(\frac{p_3}{p_2} - 1\right) \tag{32}$$

From the tests on the German aeropulse, it is found that the fuel rate for maximum thrust at $M_o = 0.515$ is 2890 lb/hr. From Fig. 4, the corrsponding pressure ratio p_3/p_2 is 5.9. Hence,

$$\text{fuel rate at } M_o = 2890 \left(\frac{1 + 0.2025M_o^2}{1.0537}\right)^{3.469} \frac{\frac{p_3}{p_2} - 1}{4.9} \tag{33}$$

The thrust is then

$$\text{thrust at } M_o = 2890 \left(\frac{1 + 0.2025M_o^2}{1.0537}\right)^{3 \cdot 469} \frac{\frac{p_3}{p_2} - 1}{4 \cdot 9} \frac{1}{S} \tag{34}$$

where s is the specific consumption in lb/hr/lb-thrust taken from the heavy curve in Fig. 4. The value of p_3/p_2 can also be taken from the same figure.

PART III

PERFORMANCE OF RAMJETS AND THEIR DESIGN PROBLEMS

By

HSUE-SHEN TSIEN

PART III

PERFORMANCE OF RAMJETS AND THEIR DESIGN PROBLEMS

DECEMBER, 1945

INTRODUCTION

The ramjet (Fig. 1) consists of a diffusor to decelerate and to compress the high velocity air stream due to the rapid forward motion, a combustion chamber for adding heat to the air mass, and a nozzle to discharge the combustion products at high velocity. The pressure in the combustion chamber, being obtained by ram compression only, is necessarily lower than that of the aeropulse moving at the same speed. In fact, at static conditions, the pressure in the combustion chamber of a ramjet is equal to the atmospheric pressure and no thrust can be produced. On the other hand, the continuous operation of the ramjet leads to a more efficient utilization of the combustion pressure. Furthermore, the inherent extreme simplicity of the power plant, as compared even with the aeropulse, is very attractive from the engineering point of view.

Compared with the turbojet, the specific fuel consumption of the ramjet is higher, especially at small flight velocities. The present available information seems to indicate that for flight velocities corresponding to a Mach number greater than 2.5, or for velocities over 1800 mph, the specific fuel consumption of the ramjet becomes comparable with that of the turbojet. Then the extreme simplicity and the lightweight of the ramjet would indicate that it is the favored power plant for such very high flight speeds, provided the air density is high enough to give a satisfactorily large thrust. For these reasons, the interest in the ramjet at present is wide-spread. However, the experimental research on this power plant is only at its beginning. The only available complete data are those obtained for the hydrogen-burning Focke-Wulf ramjet (Ref. 1). The Bumblebee project of the Bureau of Ordnance, U. S. Navy, has demonstrated the feasibility of a supersonic ramjet with a measured acceleration of the missile equal to 1 g at flight Mach number 1.5. However, no detailed performance data are available.

Historically, the concept of the ramjet as a power plant for locomotion is certainly not new. Due to its inherent simplicity, the ramjet was "invented" many times during the past decades. However, a closer examination always reveals the high fuel consumption and the inefficiency at low flight speeds in comparison with the con-

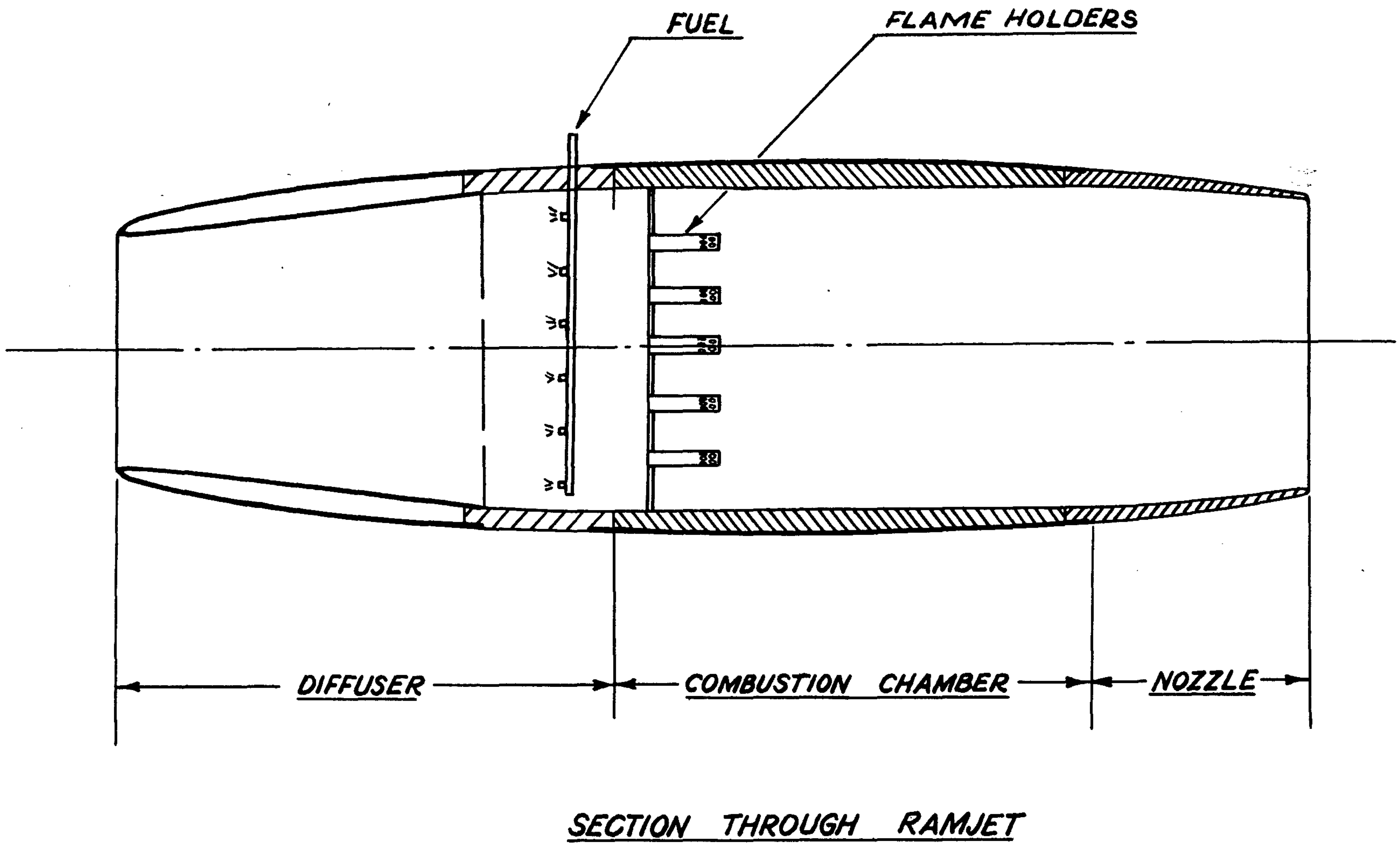


Figure 1 — Section Through Ramjet

ventional power plant such as the aircraft reciprocating engine. This is perhaps expected, as the ramjet is truly an aerodynamic engine depending upon the motion of the aircraft for efficient operation. The conventional reciprocating engine, however, is an adaptation of the stationary power plant which can operate independent of the forward motion. As the velocity of the aircraft increases, the inherent advantages of the aerodynamic power plant, the ramjet, become more evident.

During the last years of World War II, German engineers and technicians devoted a large fraction of their effort to ramjet research. Due to the lack of liquid fuel, they tried to burn coal in the form of slabs loaded directly in the combustion chamber. This is rather inconvenient, especially for long duration of operation. The Japanese started the investigation of ramjets as early as 1937. However, no fruitful result was obtained.

It is the purpose of this report to examine the general performance of the ramjet and to estimate the probable speed range for its efficient application. The discussion of performance will be based essentially on the theoretical analyses of the staff of the Jet-Propulsion Laboratory of the California Institute of Technology (Ref. 2) and by the staff of the Bureau of Aeronautics, U. S. Navy (Ref. 3). The report will conclude with a discussion of the problems of research and development, particularly the problem of combustion-chamber design.

BASIS OF THE THEORETICAL ANALYSIS

Since the ramjet is essentially a combination of diffusor, combustion chamber, and nozzle, a correct theoretical analysis should be based upon the best available information on the performance of each of these components. The essential data required are the diffusor efficiency, the combustion efficiency, the pressure drop across the combustion chamber, and the nozzle efficiency. The following is a critical evaluation of this information, and thus forms a means of judging the reliability and the accuracy of the results of the theoretical analysis.

1. *Diffusor.*

The deviation from isentropic compression in the diffusor is caused by the loss through skin friction on the wall of the diffusor, by the eddying dissipation due to boundary layer separation, and by the irreversible process of shock wave in supersonic flows. In subsonic flows, the shock loss does not occur and the losses are due to the presence of the walls of the diffusor only. Therefore, as far as the compression of the air stream through the ramjet is concerned, it is better to dispense with the diffusor completely and obtain the pressure rise by diverging streamlines ahead of the duct. This is the so-called "outside compression" as indicated in Fig. 2. However, the diverging stream ahead of the duct requires very large fairings on the outside surfaces of the duct to reduce the external drag. But even then the external drag is considerably increased. Therefore, an optimum design requires the correct compromise between the complete inside compression with a long diffusor and the complete outside compression without the diffusor. Test results of NACA (Ref. 4) show that with proper design the diffusor efficiency in subsonic flows, or the ratio of the actual pressure increase in the diffusor to the theoretical pressure increase in isentropic compression, can be made to be 85% or higher. This is the basis of Ref. 3.

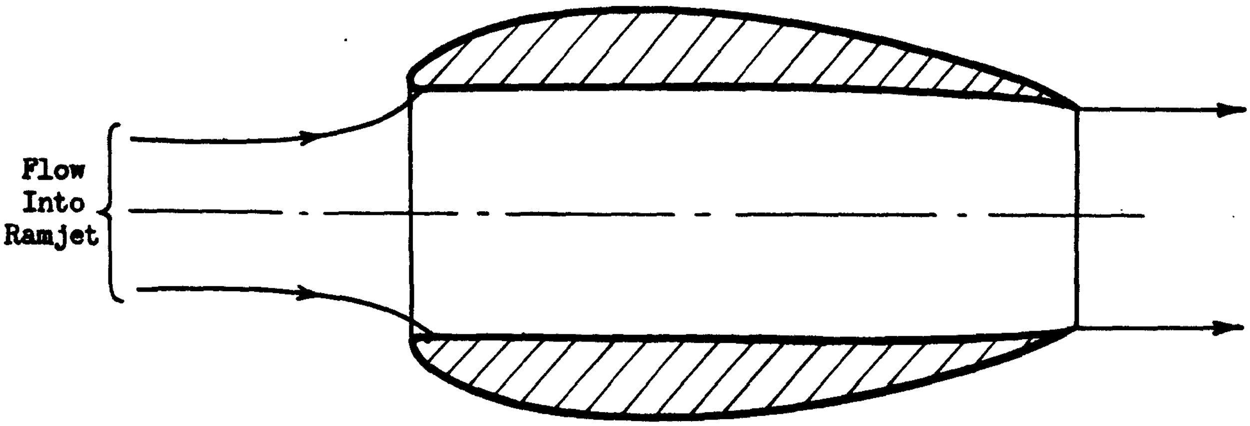


Figure 2

For supersonic flows, the matter is much more complicated due to the presence of the shock waves. If the flow through the duct is small compared with the size of the duct, then the flow velocity in the duct must be low and the shock wave is forced to the front of the diffusor opening. For this case then, the pressure rise can be calculated by assuming a normal shock from the supersonic flight velocities to subsonic velocity. After the shock, the flow is subsonic and the subsonic test data can be applied, i.e., 85% diffusor efficiency for flow behind the shock. The over-all efficiency including the shock is, however, much lower. At Mach number 3, the efficiency is only 30%. This is the basis of Ref. 2.

However, this is really an oversimplification, as the shock wave does not always occur ahead of the diffusor opening. If the flow through the duct is large, the shock wave is generally oblique to the stream direction and occurs within the diffusor. Thus, the situation is not unlike that of the diffusor after the test section of a supersonic wind tunnel. However, for the latter case, there is a rather thick boundary layer at the beginning of the diffusor while for the diffusor of the ramjet, there is no such boundary layer at the entrance. It is well known that the effect of boundary layer is to cause a premature occurrence of shock wave and thus is detrimental to the compression process. If the shock in the entrance diffusor is weaker than that in the wind-tunnel diffusor, then much better over-all diffusor efficiency than that used in Ref. 2 can be achieved. The effect is shown very clearly by the recent tests of A. Kantrowitz and C. duP. Donaldson (Ref. 5). These tests demonstrated that the efficient supersonic diffusor should be designed with a contracting entrance section before the final expanding section. Then the shock wave will occur in the neighborhood of the throat. This means that the velocity of the air stream is considerably reduced by the contracting section before the shock wave appears. Since the loss through a shock wave decreases if the velocity in front of the shock is reduced, the efficiency of such a diffusor is considerably higher than the diffusor with a normal shock ahead of the entrance as assumed in Ref. 2. Thus the diffusor compression efficiency used in Ref. 2 is on the conservative side.

Another method of improving the diffusor efficiency is to design the entrance so that a series of oblique shock waves form at the entrance section to reduce the flow velocity before entering the expanding section of the diffusor. Since the loss across an oblique shock is considerably less than that across a normal shock, the compression efficiency will be raised. K. Oswatitsch (Ref. 6) has designed such a diffusor for Mach number 2.9 with a total pressure recovery of 60% (Fig. 3). This is higher than the

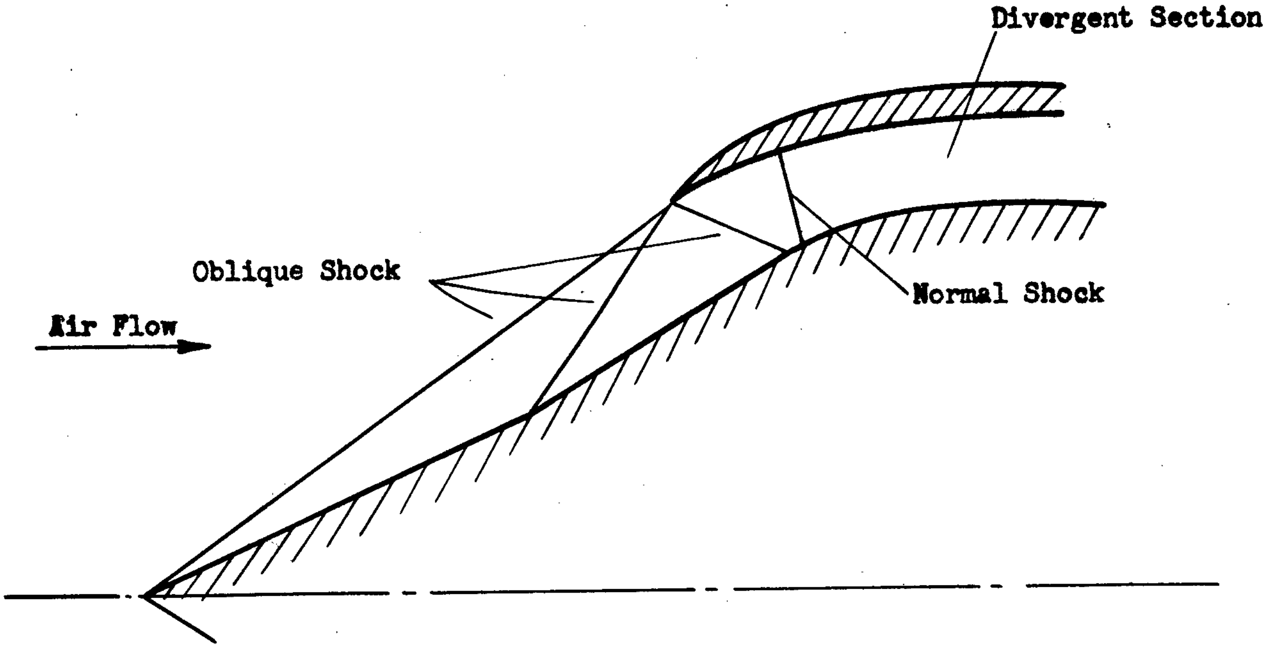


Figure 3 — Oswatitsch Diffusor

compression efficiency used in Ref. 2. The difference or improvement by these new diffusor designs is especially large at high Mach numbers. Thus the supersonic performance of a ramjet as estimated in Ref. 2 is perhaps much lower than the maximum obtainable.

2. *The Combustion Chamber.*

Due to the relatively low pressure in the ramjet combustion chamber, the thrust produced will be small unless the temperature after the combustion is raised to a high value. Furthermore, in the effort to reduce the external drag the sectional area of the combustion chamber is also reduced with a resultant high-flow velocity through the combustion chamber. These two factors of high velocity and large heat addition distinguish the ramjet combustion chamber from the turbojet combustion chamber. At present very little is known about the performance of such combustion chambers as the tests are only beginning. However, one thing is certain: The static pressure drop in the combustion chamber is a function of the degree of heating. For very large heat addition, the pressure drop is many times that of small heat addition as in the turbojet combustion chambers. The physical reason is the expansion of the gas due to high temperature. Then the velocity of flow must be greatly increased according to

the increased volume. The increase of velocity, however, can be obtained only by a large accelerating force corresponding to a large pressure drop.

The calculation in Ref. 2 takes this fact into account. The additional pressure drop due to the eddying motion created by burning and baffles is taken from preliminary test data of combustion research at the Massachusetts Institute of Technology. In the first report of Ref. 2, this additional pressure drop is assumed to be four times the dynamic pressure at the entrance to the combustion chamber. In the light of later experiments, this drop is reduced to one dynamic pressure in the second report of Ref. 2. However, the best design of ramjet combustion chamber now tested indicates that a half dynamic pressure drop is possible. Thus the calculation of Ref. 2 may be considered as somewhat conservative. On the other hand, the calculation of Ref. 3 is based upon a test on turbojet combustion chambers with low heat addition. Therefore, for large heat addition, the pressure drop must be considerably larger than the assumed value. For this reason, the value used in Ref. 3 is somewhat optimistic.

3. *Nozzle.*

Due to the nonuniformity of the flow in the cross section of the nozzle and due to skin friction, the expansion in the nozzle is not isentropic. Both calculations in Ref. 2 and in Ref. 3 use a nozzle efficiency of 95%.* This value is quite reasonable and is in agreement with the results of tests on rocket discharge nozzles.

4. *Over-all Thrust Calculation.*

In both Ref. 2 and Ref. 3, the strong interaction in subsonic flow between the jet and the flow over the outside surface of the duct is not considered. Due to the frictional forces or the forces of momentum transport between the high-speed jet and the surrounding air, the velocity over the outside surface of the duct is considerably increased. In other words, there is an effective ejector action of the jet. The net result on the performance of the ramjet is not unlike that of the thrust augmentor for the rockets. The theory of thrust augmentation is based upon the mixing of the jet and the surrounding air stream. At low velocities, the increase in thrust of the rocket system due to the augmentation can be as large as 60-80%. For the subsonic ramjet, similar augmentation exists with the resultant considerable increase in effective thrust. Physically this thrust increase comes from the stronger suction force over the leading edge of the diffusor due to increased flow velocity. The neglection of this effect in both theoretical calculations of Refs. 2 and 3 makes the results conservative. This should be kept in mind for comparing theoretical and experimental data.

what about conditions at trailing edge?

CALCULATED PERFORMANCE

The calculated performance of the ramjet as obtained in Refs. 2 and 3 will now be discussed. The results of the theoretical analyses are generally confirmed by the available test data (see Appendix).

1. *Specific Fuel Consumption.*

Since the influence of the altitude on the specific fuel consumption is small, only the result at sea-level conditions need be examined. Fig. 4 shows the comparison of

* *The definitions of the nozzle efficiency for the two calculations differ. However, the difference in the final result is not large.*

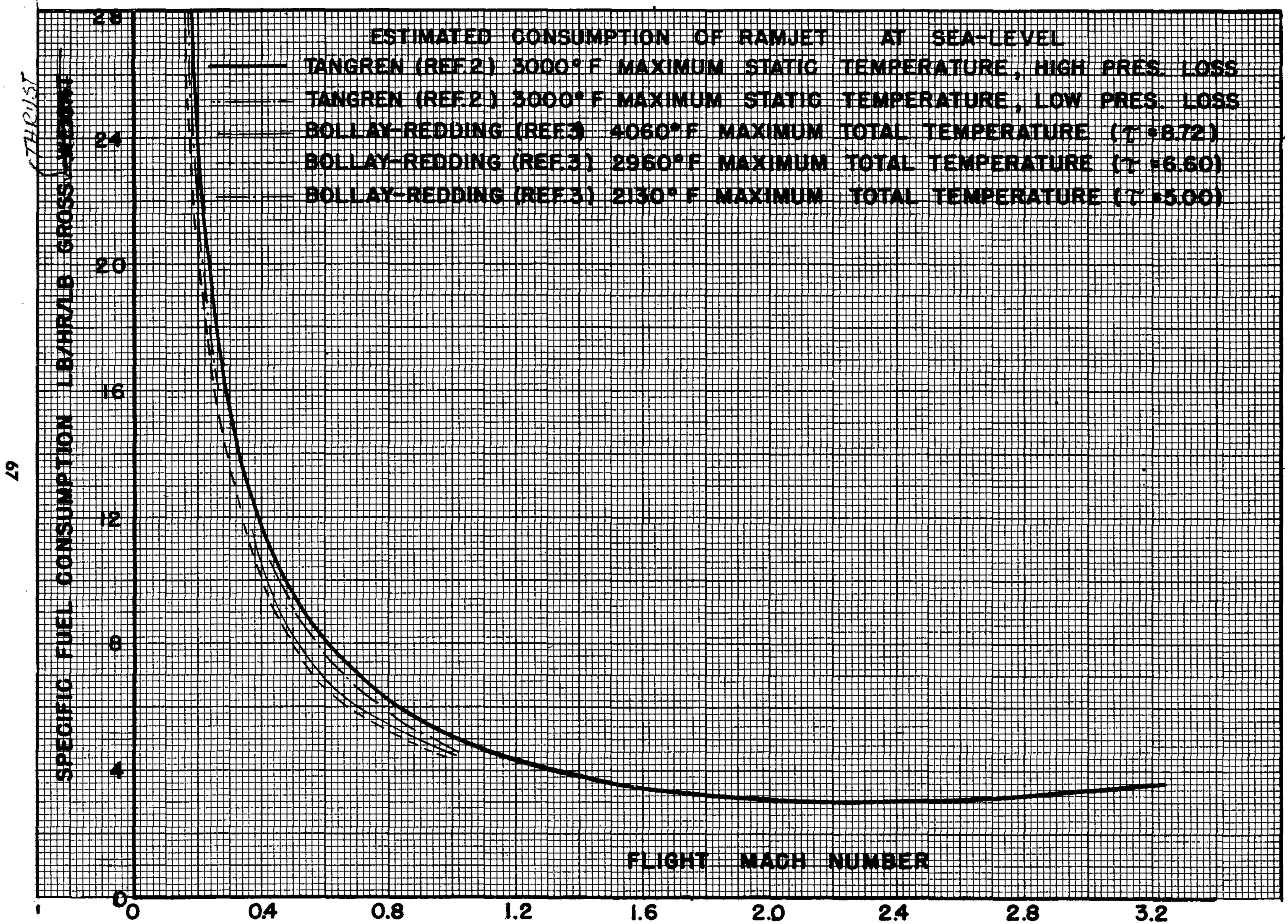


Figure 4 — Estimated Consumption of Ramjet at Sea Level

the two calculations. It is seen that due to the rather optimistic assumptions on the pressure drop of Ref. 3, the specific fuel consumption calculated is lower than that of the more conservative computations of Ref. 2. However, from the trend of the curves, this difference only occurs in subsonic flight velocities. For supersonic flight velocities, both calculations will give a specific fuel consumption of three pounds per hour per pound of gross thrust. Of course, strictly speaking, one must be careful in comparing the results, as the assumptions regarding the combustion temperatures are not the same. Ref. 2 assumes a constant static temperature at the end of combustion chamber equal to 3000°F while for Ref. 3 the ratio τ of the total temperature at the end of combustion and the atmospheric temperature is kept at a constant value for a given curve. However, the difference due to variation in τ is small and will not appreciably modify the general conclusion as stated previously.

2. *Thrust Coefficient.*

The results of calculation for sea-level conditions are again compared in Fig. 5 where the gross thrust coefficient is plotted against the flight Mach number. The thrust coefficient is defined as the ratio of thrust to the product of dynamic pressure of the free stream and the cross-sectional area of the combustion chamber. Ref. 2 assumes a constant static temperature. Therefore, more energy can be added to the gas stream as the flight velocity increases, since this additional energy is transformed directly into velocity at the exit of the combustion chamber instead of into temperature rise. The increase in the energy added is reflected in the rapid rise in thrust coefficient. However, this rapid rise is slowed down and even changes to a decrease at higher flight Mach numbers due to the choking of the combustion chamber. The choking condition is the limiting condition for stable combustion, and limits the amount of fuel that can be efficiently burned. Of course, this condition can be delayed by decreasing the inlet velocity to the combustion chamber. However, this would require an enlargement of the combustion chamber and the increase in the thrust coefficient will be small.

For subsonic velocities, the result of Ref. 2 should be compared with that of Ref. 3 for $\tau = 6.6$. Then the maximum temperature will be approximately the same. It is seen that the thrust coefficients of the two computations are nearly equal.

From Fig. 5, it is seen that the highest thrust coefficient is obtained in the range of flight Mach numbers from 1 to 2.6 for a maximum temperature of 3000°F. However, as stated in "Basis of the Theoretical Analysis," page 63, the assumed diffusor efficiency for flight Mach numbers greater than 2.0 is probably too low. Therefore, the possible performance of the ramjet at high Mach numbers will be much better than indicated by Figs. 4 and 5. In fact for Mach numbers greater than three, one has good reasons to believe that the specific fuel consumption will be close to two pounds per hour per pound thrust and the thrust coefficient will be approximately 0.8. It is then clear that the ramjet is essentially a propulsive power plant for extremely high speeds. In other words, it is an ideal power plant for guided missiles with speeds over 2000 mph if the trajectory of the missile is not too high for supplying enough atmospheric oxygen to support combustion.

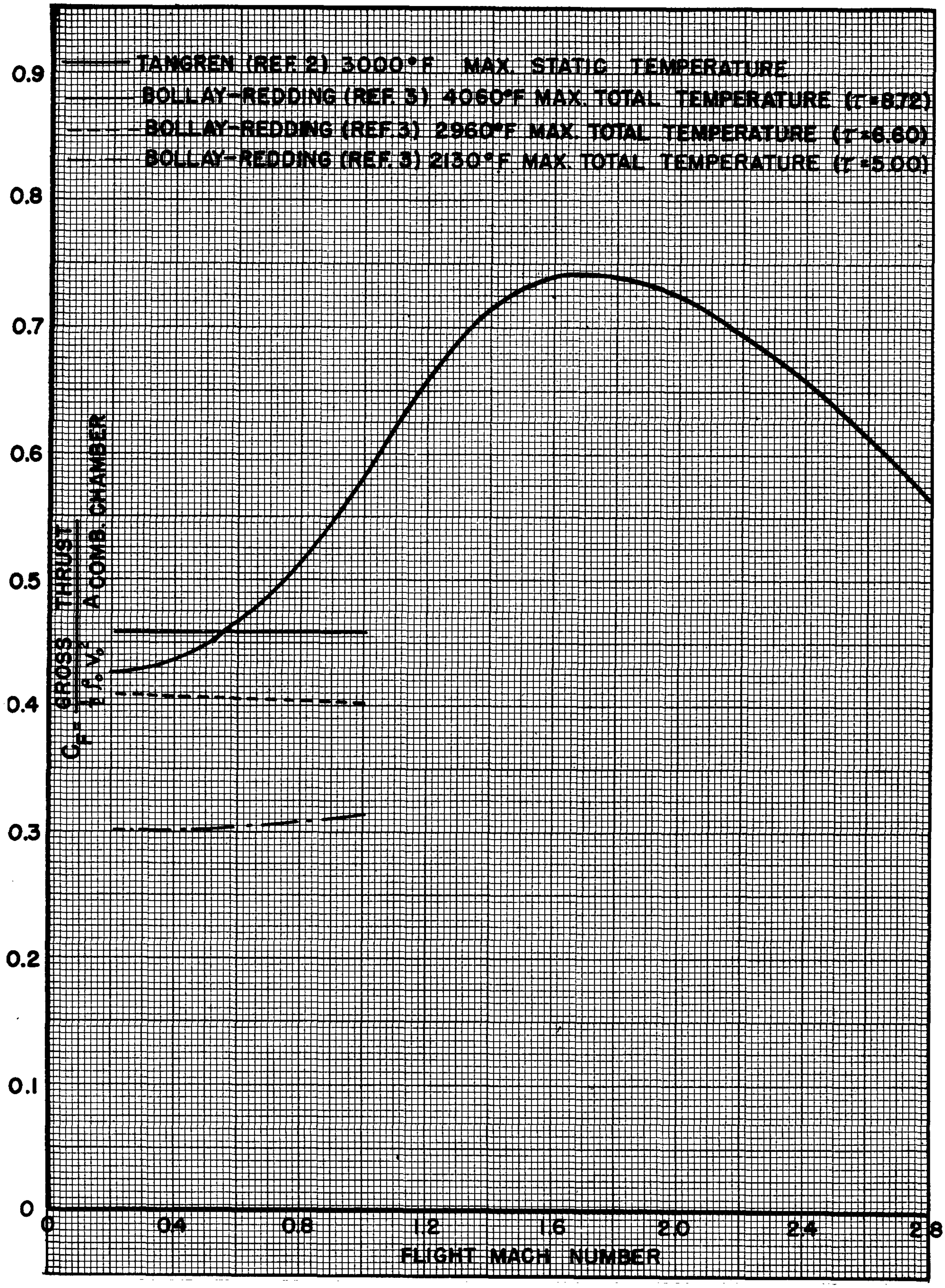


Figure 5 — Estimated Thrust of Ramjet at Sea-Level

WEIGHT OF A RAMJET

Since no actual ramjet has yet been built, it is difficult to estimate the weight of such power plants. However, due to the extreme inherent simplicity of the unit, the weight per pound of thrust should be lower than that of the aeropulse.

BOOSTING OF THE RAMJET AT LOW FORWARD SPEEDS

As stated previously, the static thrust of a ramjet is zero. In other words, the ramjet is not self-starting. Of course, the launching can be carried out by rockets until the operating speed of the ramjet is reached. A more efficient alternative is, perhaps, the ducted rocket. In this scheme, the rocket motor is installed inside the ramjet duct. The rocket jet is used to induce an air flow through the duct even at very low forward speed. Additional fuel is injected through the ramjet injector and the resultant combustion could give a large thrust. A preliminary calculation shows that at low forward speeds the thrust of the ducted rocket could be several times the value for the rocket alone. The corresponding specific consumption of both propellant and fuel is approximately six pound per hour per pound thrust. Therefore, the system is worth further analytical and experimental studies.

RESEARCH AND DEVELOPMENT ON THE RAMJET

As stated previously, the performance study of ramjets is yet in the very preliminary stage. Complete experimentation should cover a wide range of flight Mach numbers and designs for the diffusor, the combustion chamber, and the nozzle. Of course, the possibilities of boosting the thrust at low speeds should also be carefully studied. Two further specific problems of ramjet development can be stated.

1. *Aerodynamics of Power Plant Installation.*

Aside from the question of optimum design for the diffusor and nozzle, the installation of the power plant in an aircraft leads to many problems. This is true especially for the case of compressorless thermal jets which require an extremely large ratio of flow of air and thus relatively large size of the power plant compared with other components of the aircraft. Specifically, the following items should be studied: (a) the interference between the engine duct and the lifting surfaces; (b) the interaction of the hot gas jet and the flow around other components of the aircraft; (c) the possibility of combining the engine duct with the body of the aircraft, i.e., "internal installation;" and (d) the ducting of high-speed air stream from the entrance to the combustion chamber in an internal installation.

Such investigation requires a supersonic wind tunnel allowing the burning of fuels in the engine. This can be accomplished by an open-circuit tunnel or an air-exchange system. The size of the wind tunnel should be large enough to allow the use of large enough models for accurate reproduction of the prototype.

2. *High-Energy Fuels.*

If the heat value of fuels can be increased, the specific consumption will be correspondingly smaller. Here the restrictions on fuels for compressorless thermal jets are less stringent. For instance, if the combustion product contains solid particles, the

fuel will not be suitable for turbojet operation due to the possible erosive action of such particles on the turbine blades. But such fuels can be used for compressorless thermal jets. Furthermore, a small increase in the heat value has a very large beneficial effect on the pay load of the aircraft as the percentage of the fuel weight for such aircraft is necessarily very high. For instance, if the fuel weight is 70% of the total weight, while the pay load is 10% of the total weight, a 1% increase in heat value will give a 7% increase in the pay load.

Due to the high gas-stream velocity and large heat addition, the problem of combustion chamber design is very difficult. Experiments carried out by various research laboratories, especially those by the National Bureau of Standards and the Chemical Engineering Department of the Massachusetts Institute of Technology, have answered many questions. However, one must realize that such tests are exploratory rather than systematic. The fundamental problems of combustion under conditions of the ramjet are essentially aerodynamic in character. Hence the correct solution can be arrived at only by utilizing all the modern developments in fluid mechanics. This important problem will be discussed in detail in the following section.

RAMJET COMBUSTION AS A PROBLEM IN FLUID MECHANICS

The combustion if given sufficient time will reach the temperature and composition determined by chemical equilibrium. In a nonuniform mixture, there will be locally overrich regions where the combustion temperature is particularly high and the accompanying higher dissociation will reduce the efficiency of combustion. In other words, a nonuniform mixture of air and fuel will reduce the heat liberation. Therefore, for high combustion efficiency it is imperative first to obtain a uniform fuel-air mixture.

1. *Atomization of Fuel and the Mixing of Fuel and Air.*

If liquid fuel is used, the fuel-air mixture is obtained, first, through atomization of the fuel into droplets, then partial evaporation of the droplets and finally mixing of the fuel vapor with the air stream and the diffusing of droplets to form a uniform but not necessarily homegeneous fuel-air mixture. The problem involved is then essentially a diffusion and mixing problem. It is well known that diffusion and mixing in laminar flow is a very slow process, far too slow to be useful in a ramjet combustion chamber. However, if the flow is made turbulent, then the fluid elements are under rapid agitation and there is a much greater exchange between fluid elements. The diffusion and mixing in turbulent flow is thus many thousand times more rapid than in laminar flow. Therefore, turbulent flow conditions are necessary for the ramjet combustion chamber.

Modern investigations on turbulent flow show that it is not sufficient to specify just the degree of turbulence by the ratio of the fluctuating velocity to the mean velocity. One must specify the character of the turbulent flow. The character of turbulence is generally stated in terms of scales of turbulence, the correlation coefficients, the decay factor, etc. All these characteristics have a close relationship to the diffusion and mixing process. Therefore, the problem is not just to introduce turbulence into the air stream but to introduce the right kind of turbulence. In fact, even more, one has to introduce the right kind of turbulence with a minimum loss of pressure. Here the mod-

ern theory of turbulent fluid motion could be of invaluable help in solving this phase of combustion problem.

To allow for more time for the mixing of air and fuel, the liquid fuel is generally injected far ahead of the combustion zone. The recent designs of ramjets usually have the fuel injectors located in the diffusor part of the duct. The best location for uniform fuel-air mixture and necessary means of introducing the proper turbulence should be a main subject of research. However, if diffusion and mixing is the main problem here, there is no reason not to investigate the problem as such. In other words, the combustion phase of the problem can be separated from the diffusion and mixing phase of the problem. The mixture does not have to be ignited while trying to obtain a uniform fuel-air mixture. In fact, the sampling of the gas stream would be much easier in a cold test than in a hot test with combustion. One may even go another step further: If a liquid approximating the liquid fuel in its essential characteristics such as surface tension, vapor pressure, viscosity, heat of vaporization, density, etc., can be found, this liquid instead of the fuel could be used in tests, and thus possibly avoid the danger of explosion and other inconveniences. This approach to the diffusion and mixing problem would make available the whole instrumentation and methods of fluid dynamics which have been intensively developed during the past 15 years.

2. *Ignition and Flame Front,*

After having obtained a uniform fuel-air mixture, one then meets the combustion problem proper, i. e., the ignition of the mixture, and the maintenance of a stable flame with complete combustion. Generally it is recognized that to maintain a stable flame in a gas stream of very high velocity, flame holders must be used. A flame holder is nothing but a blunt body introduced into the high-speed gas stream to create a dead-water region where the velocity is low enough for a small flame to remain there. This small flame then acts as a source for continuously igniting the main high-speed gas stream. If the stream velocity is q and the flame propagation velocity is v_f, then the angle of spread β of the flame from the flame holder should be $\sin^{-1}\frac{v_f}{q} = \beta$ (Fig. 6).

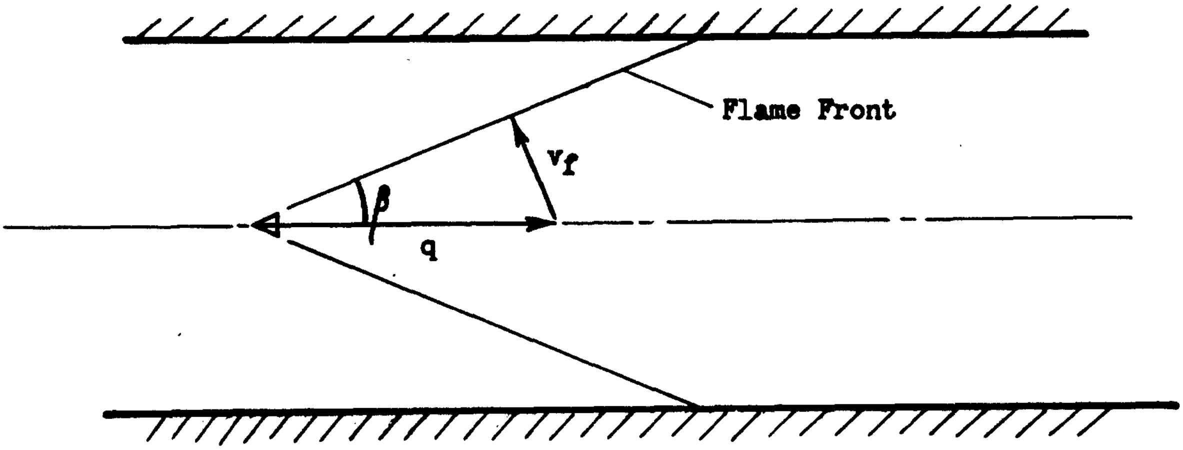


Figure 6

Therefore, to shorten the combustion chamber, we have to introduce many ignition or flame holders.

The question of flame propagation velocity is, of course, closely connected with the chemical-kinetic process of combustion. However, fluid mechanical considerations must also be introduced. First, the flow is not laminar but turbulent, i. e., in addition to molecular agitation of microscopic scale, there is also agitation of macroscopic scale with fluid elements as units. It is certain that the flame propagation velocity is a function of the scale and degree of turbulence. This relation must be thoroughly investigated. A beginning has been made by P. Chambre' and C. C. Lin (Ref. 7) in this field of research, yet much work remains to be done.

3. *Combustion in an Accelerating Gas Stream.*

After the gas mixture passes through the flame front generally the chemical reaction is not completed. It might be felt that the reaction after the flame front should be a problem purely in chemical kinetics. However, this is not the case. One must again introduce fluid mechanical consideration. First of all, the presence of turbulence greatly alters the rate of diffusion of the reacting molecules. Secondly, due to the presence of walls of the combustion chamber, generally the products of combustion cannot expand without at the same time increasing their velocity. This is the coupled effect of heating and inertia forces, i. e., aerothermodynamic effects.

To illustrate the point, consider the heating of a perfect gas in a duct of constant section. For simplicity, consider the velocity to be uniform across the section, so that a one-dimensional calculation can be made. The specific heats of the gas are assumed to be constant. Let p, ρ, v, T be the pressure, density, velocity, and temperature of the gas respectively. Then the equation of continuity is

$$\frac{d\rho}{\rho} + \frac{dv}{v} = 0 \tag{1}$$

The momentum equation is

$$\rho v dv = -dp \tag{2}$$

The energy equation is

$$dh = d\left(\frac{1}{2} v^2 + \frac{\delta}{\delta-1} \frac{P}{\rho}\right) \tag{3}$$

where dh is differential heat added per unit mass and γ is the ratio of specific heats. By simple elimination,

$$vdv = \frac{\delta-1}{\frac{1}{M^2} - 1} dh \tag{4}$$

where M is the local Mach number. Similarly, the variation of the local Mach number is

$$d(M^2) = (\delta-1) \frac{\delta M^2 - 1}{\frac{1}{M^2} - 1} \frac{dh}{a^2} \tag{5}$$

Equations (4) and (5) show that for flow velocities smaller than the local velocity of sound, both the velocity and the Mach number will increase with the addition of heat. For flow velocities greater than the local velocity of sound, both the velocity

and the Mach number will decrease with the additional heat. For these quantities to increase in supersonic flow, heat has to be subtracted from the flow i.e., make dh negative. Therefore, it is impossible to obtain supersonic velocities at the exit of a combustion chamber of constant section by continuous addition of heat if the Mach number at the inlet is smaller than one. In fact, the maximum amount of heat that can be added corresponds to that required to accelerate the gas stream to the local velocity of sound. This limit can be called the critical rich-mixture limit or simply the choking limit. This result is indeed somewhat surprising if one is limited to considerations of chemical kinetics only. It is certainly a combination of inertia and heating effect.

Now it would be natural to ask: If the fuel-air ratio is gradually increased, keeping the inlet Mach number constant, what will happen when the mixture ratio is larger than the critical value stated above? One possibility will be the flash forward rearward of the flame to the exit of the duct and combustion outside of the duct. If the flame velocity is smaller than the exit velocity of the unburned gas mixture, the flame will be "blown out." A warning before reaching this limit is perhaps rough combustion with large oscillations.

Thus it is clear that the three phases of the combustion process in ramjet, i.e., the mixing and diffusing to achieve a uniform fuel-air mixture, the ignition and flame front propagation, and finally the combustion behind the flame front, are all closely related to fluid mechanical considerations. The effect of turbulence of the gas flow and the intimately coupled effects of inertia forces and thermal energy liberated by chemical reaction must be emphasized. This point of view on the combustion problem, as a problem in the combined field of fluid mechanics and chemical kinetics, is believed to be essential for a fundamental understanding of the design principles of ramjet combustion chambers.

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APPENDIX

COMPARISON OF THEORY AND EXPERIMENT FOR A SUBSONIC RAMJET

The only complete test data on a ramjet are those obtained for the subsonic ramjet designed by Pabst of the Focke-Wulf Co. (Ref. 1). The tests were made in the A-9 wind tunnel of the Hermann Göring Research Institute at Brunswick, Germany. The diameter of the ramjet is approximately seven inches. The ramjet was supported by a streamlined strut. The drag of the strut was added to the measured thrust to obtain the net thrust of the ramjet. The fuel used was gaseous hydrogen. Since the lower heat value of gaseous hydrogen is 50,700 BTU/lb, while the lower heat value of gasoline is 19,000 BTU/lb, the measured specific fuel consumptions in hydrogen can be easily converted to those in gasoline by the mutliplying factor 50,700/19,000 = 2.67.

The reduced test data are given in Figs. 7 and 8. For the case with jet temperature equal to 1472°F, one can compare the results with the calculated values in Ref. 2 for static temperature at the exit of combustion chamber equal to 1500°F and a nozzle contraction ratio equal to 0.8. It is seen that the calculated thrust coefficient is lower than the observed value except at high speeds. The calculated specific fuel consumption is somehwat higher than the observed value except at high speeds. The calculated thrust is thus too small. It is pointed out in the section "Basis of the Theoretical Analysis," page 63, that the ejector action of the jet will tend to increase the net thrust of the unit. This seems to be confirmed by the present data. The ejector action is stronger if the jet velocity is higher or if the thrust of the unit is larger. This means higher velocity over the outer surface of the duct can be expected at a given flight Mach number if the temperature of the jet is higher. Therefore, at higher jet temperatures, the shock wave over the outer surface will appear earlier. This means that drag increase or drop in thrust coefficient due to shock wave will appear at smaller free-stream Mach number if the jet temperature is higher. This is shown by Fig. 7 to be true.

Thus the differences between theory and experiment can be easily explained. The trend and magnitude of the specific consumptions and the thrust coefficients are, however, satisfactorily predicted by the theory. Hence the general validity of the assumptions made in the theoretical analyses is established.

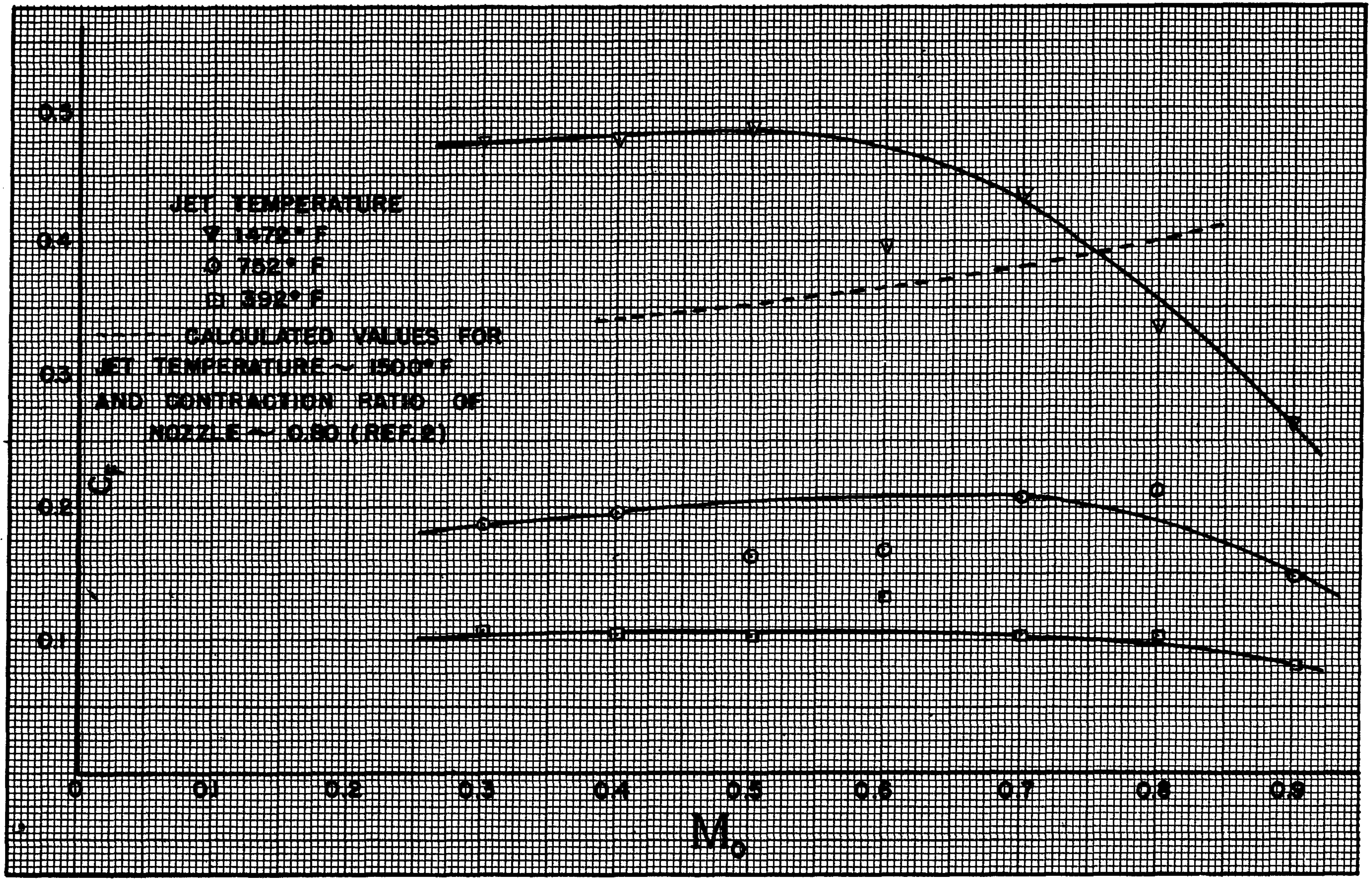

Figure 7 — Calculated Values for Jet Temperature — 1500°F and Contraction Ratio of Nozzle — 0.80 (Ref. 2)

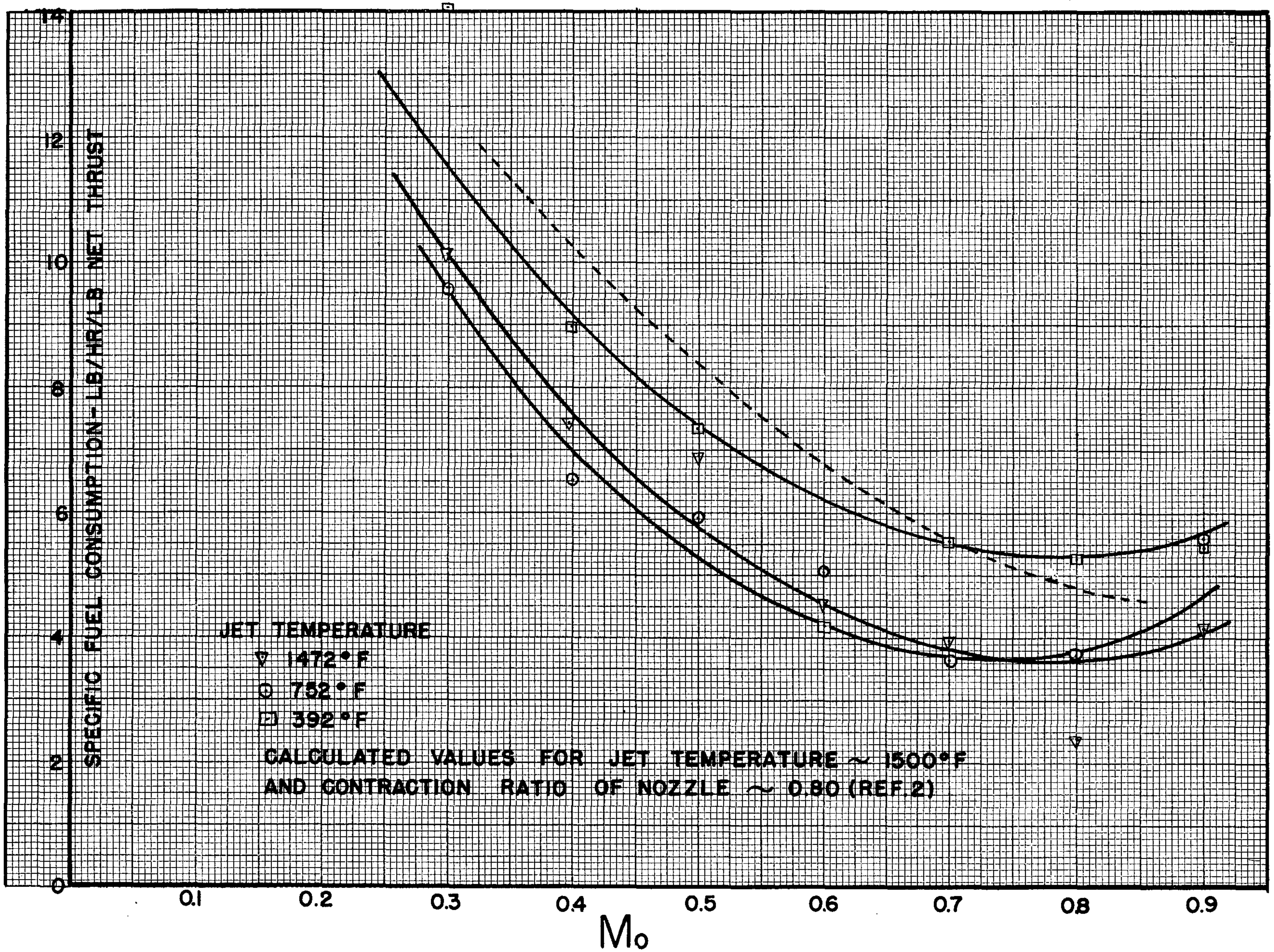

Figure 8 — Calculated Values for Jet Temperature — 1500°F and Contraction Ratio of Nozzle — 0.80 (Ref. 2)

PART IV

FUTURE TRENDS IN THE DESIGN AND DEVELOPMENT OF SOLID AND LIQUID FUEL ROCKETS

By

HSUE-SHEN TSIEN

PART IV

FUTURE TRENDS IN THE DESIGN AND DEVELOPMENT OF SOLID AND LIQUID FUEL ROCKETS*

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TYPES OF ROCKETS AND THEIR PRESENT APPLICATIONS

The two main types of rockets are the solid-propellant type and the liquid-propellant type. The solid-propellant rockets are now used or suggested for use in propelling artillery rockets, for the assisted take-off of aircraft, for the launching of flying bombs and missiles, and for the propulsion of large missiles. The liquid-propellant rockets are used for assisted take-off and for the propulsion of very large missiles and airplanes. While there is no essential difference in the operating characteristics of these two types of rockets, and thus for any new application, the possibility of both types should be investigated, there are certain facts which should be kept in mind. The solid-propellant rocket contains all the propellant in the high-pressure combustion chamber or the motor. Thus if the duration of the operation is long, the chamber volume becomes very large and the weight of the chamber will be very large. Therefore, for very long durations, i.e., durations in excess of 30 or 40 sec, the weight of a solid-propellant rocket is heavier than that of a liquid-propellant rocket. However, this line of demarcation also depends upon the thrust of the rocket. The reason for this variation is that for the liquid-propellant rocket the unit weight is a function of the thrust. Larger thrust makes the unit weight smaller, especially in the case of pump-fed rockets. The preceding value of 30 or 40 sec corresponds to a thrust of approximately 4000 lb. In other words, for durations in excess of 30 or 40 sec and for thrust in excess of 4000 lb, the liquid-propellant rockets are definitely superior to the solid-propellant rockets. For durations or thrust below these limits, generally one can say that for short durations at large thrust, such as artillery rockets, solid-propellant rockets are more suitable; while for long durations at a small thrust, as in airplane propulsion, the liquid-propellant rockets are superior.

* In order to cover the whole field of rocket development, liberal use has been made of material discussed in greater detail in the two papers, L. P. Hammett, "Solid Propellants for Rockets and other Jet-Propelled Devices;" A. J. Stosick, "Liquid Propellants for Rocket-Type Motors," in the volume *Aircraft Fuels and Propellants,* a further report of the Scientific Advisory Group.

SOLID PROPELLANT ROCKETS

1. *Present Status of the Solid Propellant Rockets.*

Solid propellants are often discussed from the point of view of thermal energy, which is essentially connected with the effective exhaust velocity or the specific fuel consumption. However, there are many other characteristics of solid propellants which greatly influence their usefulness as rocket propellants. These characteristics are as given in Table I: (a) density, (b) temperature sensitivity, (c) rate of burning, and (d) exponents in the rate of burning law.

The density of the propellant is important in that it determines the volume of the combustion chamber if the specific impulses of the propellant remain constant. In other words, for a given weight a less dense propellant requires a larger combustion chamber than a denser propellant. Since for solid-propellant rockets the main part of the motor weight is the weight of the combustion chamber, a denser propellant will lead to a lighter unit. For most solid propellants, the density is 100 lb /cu ft.

The temperature sensitivity is really the sensitivity of the burning rate to the temperature of the propellant before ignition, i.e., the ambient temperature. A high sensitivity would mean a large variation of the chamber pressure for a given motor design. To operate the rocket at high ambient temperature, the chamber has to be designed to withstand the high pressure, and this leads to the heavy motor weight. At low ambient temperature, the pressure is too low for stable combustion. Therefore, if the temperature sensitivity is high, the rocket motor operating temperature range will be limited. The double-base propellant ballistite is an example of high-sensitivity propellant.

The rate of burning of the propellant determines the burning surface. It is seen from Table I that this rate varies but little. The large difference in the operating duration from 0.5 seconds for the artillery rockets to 30 seconds for assisted-take-off rockets is achieved by the difference in design of the propellant charge. For short burning time, the charge is designed in such a manner that it will burn with large surface, and thus the volume or weight of charge burned per second is large. This is the so-called unrestricted burning. The other type for long burning time is the so-called restricted burning. In this case, the burning surface is generally restricted to the cross-sectional area of the combustion chamber. It is thus immediately clear that for a restricted-burning rocket, the motor could be loaded to a fuller extent, since gas passage ducts are not required in the motor, and thus the motor will be more compact and lighter in weight. From this point of view, then, a restricted-burning rocket is preferred. This is clearly seen in Table II, where the performance of the actual solid-propellant rockets is compared. The impulse to total weight ratios for the unrestricted-burning motors are lower than those for the restricted-burning rockets.

The rate of burning of a solid propellant varies as a power n with the pressure of the gas surrounding it, while the flow of the gas through the nozzle varies as the first power of the pressure. This situation admits of a stable steady state only if n is less than unity. The state is more stable and less sensitive to disturbing influences, such as variations in the burning surface, the greater the value of $\frac{1}{n}$. In this sense the difference between the exponent 0.75 of most double-base propellants and the 0.4–0.5 exhibited

TABLE I

CHARACTERISTICS OF SEVERAL SOLID PROPELLANTS

	Ballistite	*Composite Propellants*		*GALCIT 53 (Asphaltic)*	*GALCIT 58 (Asphaltic)*	*GALCIT 61C (Asphaltic)*	*GALCIT 65 (Asphaltic)*
		Slow	*Fast*				
Specific Impulse, lb-sec/lb	210	160	170	170	174	186	177
Exhaust Velocity, ft/sec	6800	5150	5500	5500	5600	5900	5700
Density, lb/ft^3	101.5	101	112	104	108	110	110
Impulse per Unit Volume, percent	100	76	90	83	89	96	92
Temperature Sensitivity, percent increase in chamber pressure between 40° and 90°F	29	10	14	10	10	11	11
Temperature Range for Safe Operation, °F (Laboratory Tests)	Limited by Sensitivity	−40 to 140	−40 to 140	30 to 120	15 to 120	−9 to 120	−5 to 120
Rate of Burning, in./sec	1.40	0.25	1.32	1.32	1.80	1.60	1.46
Exponent in Rate of Burning	0.69	0.40	0.40	0.69	0.80	0.76	0.7
Chamber Temperature, °F	5000 to 6000	3000 to 3500	3000 to 3500	3000 to 3500	3000 to 3500	3000 to 3500	3000 to 3500

Above data are all for 2000 psi chamber pressure, 14.7 psi external pressure.

by composite propellants, is a large and important one, which leads to materially greater reproducibility and reliability with the latter material.

2. *Possible Improvements in Performance of Solid-Propellant Rockets.*

The possibility of raising the performance of solid-propellant rockets by improving the energy of the propellant is not oustanding. However, by considering the matter in its broad outline, it is evident that the aim of rocket design is to raise the impulse produced per unit total weight, i. e., the weight of the propellant plus the weight of the motor. Therefore, one can achieve this by either decreasing the propellant weight or by decreasing the motor weight. These possibilities are clearly shown for the particular application of artillery rockets in Figs. 1 and 2. The effect of variations of the effective exhaust velocity and the motor weight on the maximum flight velocity of the rocket, while keeping the pay load constant, is shown in Fig. 1. The effect of variations of the effective exhaust velocity and the motor weight on the pay load, while keeping the maximum flight velocity constant, is shown in Fig. 2. It is seen that almost the same percentage gain can be achieved by decreasing the motor weight as by increasing the effective exhaust velocity. Hence the design factors are just as important in improving the performance of solid propellants as is the propellant itself.

In the following paragraphs, the different aspects of the lightweight design will be discussed in detail.

a. LOW-COMBUSTION PRESSURE. Since the weight of a rocket motor increases approximately in direct proportion to the gas pressure, while the effective exhaust velocity varies much less rapidly, a decrease in operating pressure is favorable until it reaches the point where other forces than bursting pressure become important factors in motor design, certainly as far down as 500 psi. In general, solid propellants possess a lower pressure limit, which depends on size and geometry of motor as well as on the propellant, below which they fail to burn, or burn irregularly and incompletely. This is shown in Fig. 3, which gives the chamber pressure record of the burning of a double-base propellant at low pressure. The normal burning pressure of this propellant is approximately 1000 psi. The irregularity at low pressure is evident. The lowest pressure for smooth burning is called the lower pressure limit. Obviously a low value of this limit is a desirable property of a propellant, and possibilities in this direction should be exploited.

The irregular burning at low combustion pressure can also be controlled by a spring-loaded control valve, as shown in Fig. 4. It is the German practice to use one of the valves with one or two conventional discharge nozzles, so that when the gas pressure in the chamber is higher than the preset value, the valve is opened and the pressure is released, and when the gas pressure in the chamber is lower than the preset value, the valve is closed and the pressure in the chamber is built up. The smooth combustion with the control valve is shown in Fig. 5 as compared with the rough and intermittent burning of the same propellant without the valve shown in Fig. 3. Of course, the addition of the control valve is a mechanical complication. However, the possibility of lowering the combustion pressure and the consequent saving in weight, especially for long-duration rockets, may well overbalance the inconvenience.

b. SMALL EXPONENT IN BURNING LAW. The effect of a small exponent in the burning law on stability and reproducibility has already been pointed out in a

TABLE II

PERFORMANCE OF SOLID PROPELLANT ROCKETS

	Type	*Total Weight, lb*	*Impulse, lb/sec*	*Impulse Weight Ratio*	*Weight of Propellant*
Unrestricted Burning	3.5 in. AR-M5	33.8	1,820	53.6	8.5
	5 in. AR-M7	33.8	1,800	53.2	8.5
	5 in. HVAR	88.4	5,320	61.0	24.0
	11.75 in. AR-M3	690.0	32,900	47.6	148
	7.2 in. T21	20.54	877	42.6	5.74
Restricted Burning (Aerojet Co.)	X8AS-1000	150	12,500	83.3	69
	X10AS-1000	200	16,000	80.0	90
	12AS-1000	250	20,000	80.0	108
	30AS-1000	415	36,000	86.7	192

Figure 1

Effect of changes in effective gas velocity, in propellant weight, and in motor weight, on velocity of 5.0 in. HVAR. Ordinates are rocket velocities.

Curve I. Effect of variation in effective gas velocity over range from 50 to 150% of value for existing rocket (7130 ft/sec). All weights constant.

Curve II. Effect of variation in weight of propellant over range from 50 to 150% of value for existing rocket (24.0 lb). Effective gas velocity (7130 ft/sec), pay load (48.2 lb), and motor weight (64.4 lb) constant.

Curve III. Effect of variation in weight of motor over range from 50 to 150% of value for existing rocket (64.4 lb). Effective gas velocity (7130 ft/sec), pay load 48.2 lb), and propellant weight (24.0 lb) constant.

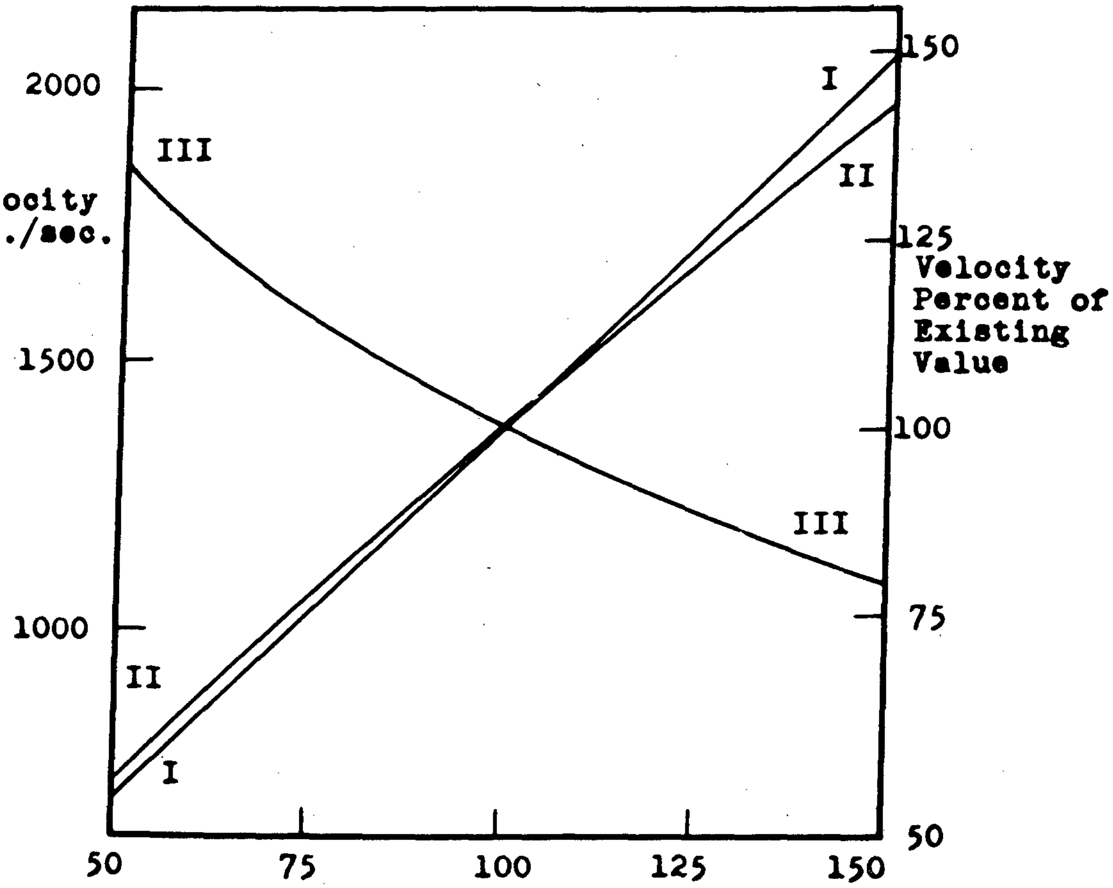


Figure 1 — Percent of Value for Existing Rocket

Figure 2

Effect of changes in effective gas velocity, in propellant weight, and in motor weight on pay load of 5.0 in. HVAR. Ordinates are values of pay load.

Curve I. Effect of variation in effective gas velocity over range from 50 to 150% of value for existing 5.0 in. HVAR (7130 ft/sec). Motor weight (64.4 lb), propellant weight (24.0 lb), and projectile velocity constant (1375 ft/sec).

Curve II. Effect of variation in weight of propellant over range from 50 to 150% of value for existing rocket (24.0 lb). Effective gas velocity (7130 ft/sec), projectile velocity (1375 ft/sec), and motor weight (64.4 lb) constant.

Curve III. Effect of variation in weight of motor over range from 50 to 150% of value for existing rocket (64.4 lb). Effective gas velocity (7130 ft/sec), projectile velocity (1375 ft/sec), and propellant weight (24.0 lb) constant.

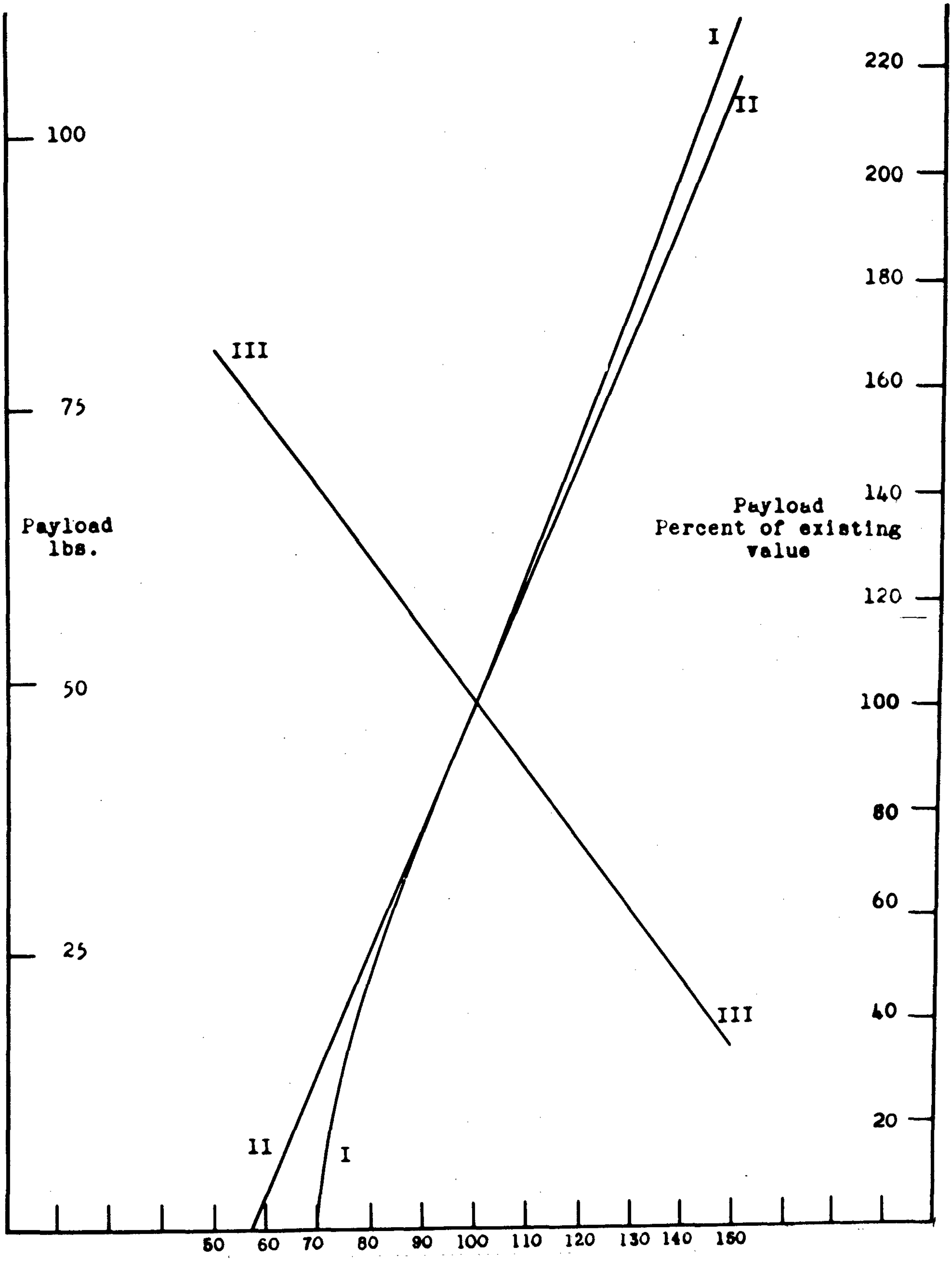


Figure 2 — Percent of Value for Existing Rocket

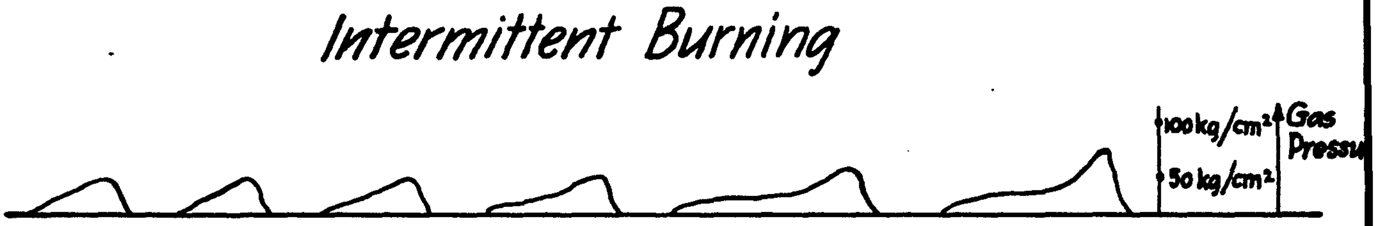


Figure 3 — Intermittent Burning

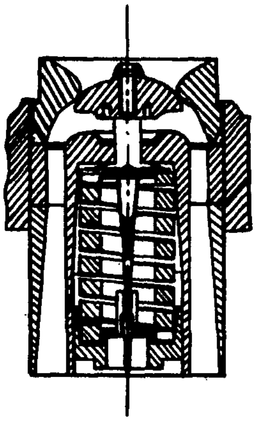


Figure 4

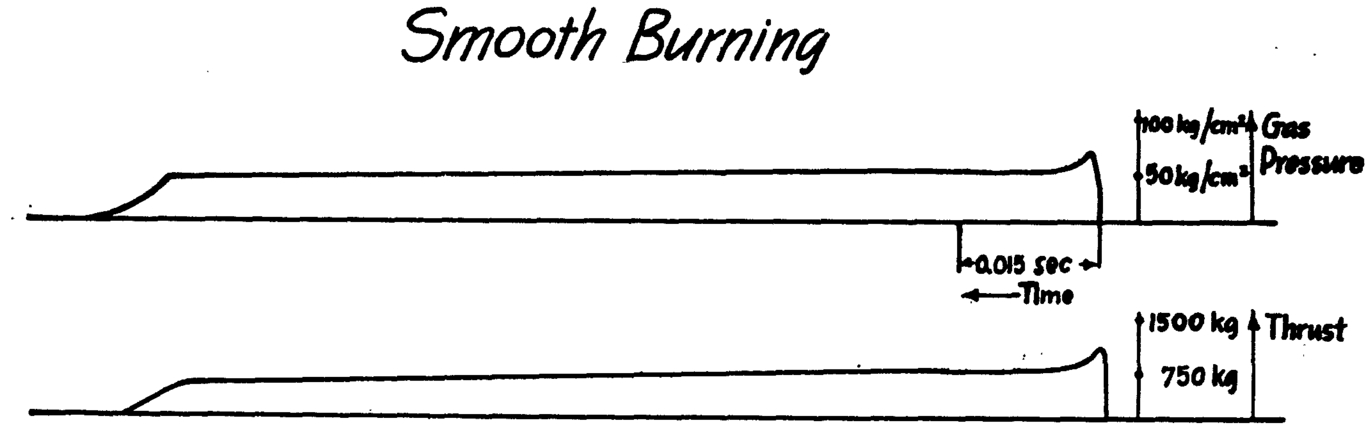


Figure 5 — Smooth Burning

previous paragraph. The advantage also allows the designer to use a lower factor of safety in the chamber design. This naturally leads to a lighter motor.

The value of a low exponent is especially great in the usual type of artillery rocket, in which a tubular or cruciform grain burning over all or nearly all of its surface is contained in a long motor of small diameter. When an attempt is made to reach the highest possible ratio of propellant weight to motor weight, the limiting factor becomes the space available for flow of gases toward the nozzle. If this is small, a pressure differential is set up, the burning rate increases in the high-pressure region, and the whole effect is magnified by an amount which is greater, the greater the exponent. Consequently, a low exponent permits a higher propellant-motor weight ratio for a specified time of burning. A rigid propellant is of course desirable for the same reason, as is one resistant to distortion under applied forces, since any bulging of the grain under the forces of setback and pressure drop reduces the port area.

c. LOW-TEMPERATURE SENSITIVITY. As stated before, the temperature sensitivity is closely related to the permissible range of operating ambient temperature. A low temperature sensitivity is desired. Such a characteristic is also desirable from the point of view of low motor weight, as the advantage of operating at a low pressure vanishes if the pressure becomes four or five times greater at temperatures in the upper part of the operating range, due to high temperature sensitivity.

It has been suggested by L. Pauling that a charge containing particles or strands of small diameter, composed of a fast-burning powder of low temperature coefficient, embedded in a matrix of slower burning powder of high temperature coefficient, possesses, as a whole, the low coefficient of the particles. In this case the grain does not burn by recession of a plane surface, but by the formation of a broken surface whose magnitude is determined by the behavior of the faster burning powder. This procedure, already proven experimentally, makes it possible to eliminate the undesirably high temperature coefficient of the double-base propellant by the addition of a proportion of solvent-extended composite propellant.

d. RESTRICTED BURNING AND HIGH BURNING RATE. As mentioned previously, the restricted-burning solid-propellant rockets are superior to the unrestricted-burning rockets, due to better charge-loading efficiency. However, to make a restricted-burning rocket for very short duration of burning, one has to develop a propellant or a group of propellants with a much higher burning rate than those available today. It seems that by addition of certain metallic powders, such as in the Aberdeen propellant, this can be achieved. The actual research work on this type of solid propellant is, however, just beginning, and the final satisfactory product is still far away. As a tentative aim, one might set the limits of burning rate from one to 100 inches per second. Of course, to develop these propellants, a closer understanding of the combustion phenomema is necessary. Such understanding will not only greatly facilitate the research work on burning rates, but also will be very useful in the reduction of the temperature sensitivity of the propellant. It is generally true that the temperature sensitivity is higher for unrestricted burning than for restricted burning for the same propellant. Thus development of a propellant of high burning rate, so that the burning can be restricted to the cross-sectional area of the motor, is also desirable from the point of view of temperature sensitivity.

e. OTHER POSSIBLE IMPROVEMENTS. To reduce the weight of the motor, one can also try to reduce the temperature to which the motor walls are heated by the propellant gases, since the strength of metals decreases materially at temperatures easily reached by the motor walls in conventional designs. Since the propellant itself is a very good insulator, for a short-duration rocket, the charge might be made to burn from a central perforation along the axis of the charge and gradually to burn toward the wall of the motor. If this can be done, then aluminum, magnesium, or even plastic walls might be used. A further possibility is the ceramic-coated wall, to insulate the hot gas from the metal itself.

Smoke elimination is sometimes desirable, as for ground-fired rockets, but is unimportant for others, such as airborne missiles. The same considerations apply to flash, i. e., a luminous jet.

3. *Concluding Remarks.*

In this section, the status of the present solid-propellant rocket development is briefly reviewed. The various possibilities of improving the performance, such as low temperature sensitivity, low combustion pressure, and small exponent in burning law are discussed. It is believed that the solution of these problems will greatly increase the usefulness of the solid-propellant rockets, which possess the advantage of inherent simplicity.

LIQUID PROPELLANT ROCKETS

1. *Design Criteria for Liquid-Propellant Rockets and Choice of Propellants.*

The two main applications for liquid-propellant rockets are the assisted take-off of large aircraft, when a thrust larger than 6000 pounds is required for durations up to 40 seconds, and the propulsion of large guided missiles, which usually need thrust in excess of 20,000 pounds for durations up to 80 seconds. The design criteria for these two main applications are not the same. For the assisted-take-off application, the essential requirements are (a) reliability of operation, (b) simplicity of servicing, and (c) long life for repeated application. On the other hand, the problem of fuel consumption is not very acute, as the weight of the rocket propellant is only a very small fraction of the take-off weight of the aircraft, and since it is consumed at the end of the take-off process, it does not constitute a load in the normal operation of the airplane. For the missile application, the first requirement is high specific impulse for range. Simplicity of design and ease of handling are secondary. On the other hand, if the missile is intended to be used on a very large scale, the aspect of production economy must also be considered.

Table III gives the calculated value or the expected value of the performance of the different liquid propellants. The experimental values are generally lower by ten percent than the calculated values, due to unavoidable losses. The nitromethane propellant and the hydrogen-peroxide propellant are outstanding in that the chamber temperature is low. This fact, together with the advantage of having to deal with only a single component, naturally leads to their adaptation for the assisted take-off of aircraft. Nitromethane itself is somewhat shock sensitive. But this sensitivity can be considerably reduced by the addition of a few percent of a desensitizer, such as methyl

TABLE III
CALCULATED PERFORMANCE OF VARIOUS LIQUID PROPELLANTS

(at 300 psi chamber pressure)

Propellant	*Mixture Ratio*	*Exhaust Velocity ft/sec*	*Specific Impulse lb-sec/lb*	*Volumetric Specific Impulse lb-sec/ft³*	*Chamber Temperature °F*	*Exit Temperature °F*	*Ratio of Specific Heats of Products*	*Average Molecular Weight of Products*
Liquid Oxygen, Gasoline	2.5	7,780	242	14,958	5,470	2,990	1.219	22.66
Liquid Oxygen Ethyl Alcohol	1.5	7,830	244	14,777	5,260	2,965	1.212	21.69
Liquid Oxygen Methyl Alcohol	1.0	7,587	236	13,702	4,760	....	....	....
Liquid Oxygen, Ammonia	1.4	8,000	248	13,161	4,950	....	....	....
Liquid Oxygen, Hydrazine	0.4	8,285	257	17,327	4,120	....	....	....
Red Fuming Nitric Acid Aniline	3.0	7,091	221	18,763	5,065	2,746	1.220	25.40
White Fuming Nitric Acid Aniline	3.0	7,035	219	18,184	4,900	2,635	1.221	25.01
White Fuming Nitric Acid Furfuryl Alcohol	2.0	6,840	212	17,867	4,293	2,578	....	....
Mixed Acid Monoethylaniline	2.5	6,780	210	17,569	4,600	2,800	....	....
Nitromethane	....	7,008	218	15,515	3,950	1,980	1.245	20.31
Diethyleneglycol, Dinitrate	....	6,865	213	18,484	4,078	2,061	1.236	21.79
Hydrogen Peroxide (100%)	....	4,710	146	13,184	1,794	714	1.249	22.70
Hydrogen Peroxide (87%)	....	4,065	126	11,039	1,216	379	1.271	21.92
Hydrogen Peroxide (87%) Methyl Alcohol	4.0	7,180	223	16,845	4,156	2,538	....	....
Hydrogen Peroxide (87%) Nitromethane with 30% Methyl Alcohol	1.0	7,305	227	17,007	4,050	....	....	....
Hydrogen Peroxide (87%) Nitromethane with 35% Nitroethane	0.3	7,014	218	15,788	2,864	1,966	....	....
Hydrogen Peroxide (87%) Hydrazine Hydrate	1.6	7,009	218	16,468	4,000	1,950	....	....

alcohol. Hydrogen-peroxide propellant has the lowest chamber temperature. In fact, the chamber temperature is so low (1220°F) that no cooling of the motor is required. This is a great advantage. On the other hand, the inflammability of concentrated hydrogen peroxide with any organic matter, such as cellulose, causes difficulties in the handling of the propellant, although these difficulties seem to have been overcome by the Germans. A second choice of propellant for the aircraft application is a bipropellant, such as hydrogen peroxide plus methyl alcohol, or nitric acid plus aniline. While these propellants have good performance, the two components complicate the feed system. The higher combustion temperature also makes the cooling of the motor more difficult. Although the performance of the propellant with red fuming nitric acid is better than the performance of the propellant with white nitric acid plus a small percentage of catalyzer, the nitric oxide fumes (NO_2) are objectionable for the aircraft applications and the latter type is preferred.

For the long-range missile application, the most effective propellants are those with liquid oxygen as the oxidizer. However, due to the difficulty of using liquid oxygen as a coolant for the motor, which is subjected to very high heat flow due to high combustion temperature, it is preferable to have fuel in the liquid state under the conditions of temperature and pressure in the cooling ducts, and of such composition as to make adequate cooling possible. Thus the combinations should be liquid oxygen plus ethyl alcohol, liquid oxygen plus methyl alcohol, or liquid oxygen plus hydrazine. The last-mentioned combination is new and has not yet been tested. However, it gives the highest performance and holds much promise.

The density of the propellant should also be considered in conjunction with the specific impulse. Liquid hydrogen is rejected as a possible fuel on the ground of its extremely low density, which necessitates large tank volume for storage. This fact is particularly important for antiaircraft guided missiles, which must travel at high speeds in relatively dense air. Hence the air resistance associated with the large body required by the low-density propellant is a distinct disadvantage. This is demonstrated in Table IV, where three winged missiles of the same explosive load, and with three different propellants, are compared. The nitric acid and aniline propellant gives about the same flight duration as other higher preformance propellants. This fact, together with the ease of manufacture and the spontaneous inflammability, led the Germans to adopt this type of propellant for their antiaircraft rockets developed during the last years of World War II. One disadvantage of the large-scale use of the

TABLE IV

Propellant	*Liquid* O_2+ *Liquid* H_2	*Liquid* O_2+ CH_3OH	HNO_3+ *Aniline*
Structure Weight and Pay Load, lb	3340	3130	3100
Propellant Weight, lb	3700	4770	5260
Gross Weight, lb	7040	7900	8360
Specific Gravity of Propellant	0.4	1	1.4
Specific Consumption, lb/hr/lb-thrust	12.6	16.2	18
Volume of Propellant Tanks, cu ft	150	77	61
Body Frontal Area, sq ft	19.4	12.9	10.8
Wing Area, sq ft	161	183	194
Flight Duration, min	19.5	20.5	19.5

nitric acid plus aniline propellant is, perhaps, the high cost of the aniline. However, the German discovery of the possibility of adding other cheap inert fuels and then retaining the spontaneous inflammability by use of a catalyzer is most interesting and should be exploited. The use of a mixture for the fuel also offers a means of controlling other properties of the propellant, such as freezing point and viscosity, for wide operating temperature range of the rocket.

2. *Propellant Feed Systems.*

a. COMPARISON OF VARIOUS FEED SYSTEMS. Since the propellant of a liquid-propellant rocket is stored in tanks, a feed system is necessary to force the propellant from the tank to the combustion chamber, under high pressure. The various systems developed to date (listed in order of decreasing weight for a given amount of propellant and chamber pressure) are pressurized-gas system, gas-generator system, and the turbine-pump system. In the pressurized-gas system, an inert gas such as nitrogen or air is stored under very high pressure (up to 2000 psi). The gas is allowed to expand through a pressure regulator and to act on the propellants in their tanks to force them through the propellant injector to the combustion chamber. The propellant tanks are thus under high pressure and have to be built strong and heavy. The weight of the inert gas and the high-pressure gas tank is proportional to the volume of the propellant to be injected.

In the gas-generator system, the required higy-pressure gas is produced not from a high-pressure gas tank, but from either the burning of a solid charge or the burning of a part of the main propellant. Since the high-pressure gas is produced continuously, the apparatus is much smaller and weighs much less. If the gas is produced by combustion of a part of the main propellant, the size and weight of the gas producer is independent of the total amount of the propellant to be injected, but is directly proportional to the rate at which the propellant is injected. Although this system still requires pressurized propellant tanks, the weight of a large high-pressure gas tank of the pressurized-gas system is eliminated. Consequently the weight of the whole system is less.

The lightest of the propellant-feed systems is the turbine-pump system. Here the propellant tanks can be built light, as they are not under pressure. The turbine is driven by hot gas obtained from the combustion of a small amount of the liquid propellant. The propellant is transferred from the tanks to the chamber by pumps driven by the turbine. At present the only type of pump used is the centrifugal pump. These pumps have the advantage of small dimensions and hence light weight. However, centrifugal pumps cannot be operated at very high efficiency if the propellant flow is small and the feed pressure high. Thus, other types of fluid pumps should also be explored, especially with regard to the possible improvement in pumping fluids of high vapor pressure, such as fuming nitric acid and liquid oxygen. The weight of the turbine-pump aggregate is proportional to the rate of propellant flow, but independent of the total amount to be pumped.

Thus the essential difference between the pressurized-gas system and the pump system is that while the weight of the former is proportional to the total quantity of the propellant, the weight of the latter is more closely related to the rate of propellant

flow. Therefore, for short-duration large-thrust operation, the pressurized-gas system is definitely superior. But for long-duration operation, the pump-feed system is better. For a duration of 60 seconds and a thrust of 4000 pounds, the pump-feed system is the lightest. However, the pump system suffers from the complexity of the machinery and hence higher manufacturing cost. For an expendable weapon such as a guided missile, where large-scale application is envisaged, the simplicity of the gas-generating system may have many advantages when the man-hours required for its production are considered. Thus the two feed systems that should be intensively developed are the gas-generating system and the turbine-pump system.

b. PUMP DRIVES. For many applications of the liquid-propellant rockets, the power required to drive the pumps can also be derived from other sources. The possible sources are: (1) mechanically connected with an auxiliary engine or with the main power plant of the aircraft, if the rocket is only used for auxiliary propulsion; (2) separate gas turbine driven by rocket propellant; (3) rotary rocket motors; and (4) windmill.

For assisted take-off of aircraft, or for superperformance of aircraft, the pumping power can be taken most conveniently from the main power plant of the aircraft. However, two facts must be kept in mind. The first is that for the turbojet power plant there is a delicate balance between the gas-turbine power output and the compressor power absorbed. If a certain amount of gas-turbine power output is bled off for driving the feed pumps, proper measures, such as variable discharge nozzle, must be taken to insure the satisfactory operation of the turbojet system, i.e., to maintain the matching of turbine and compressor with the pump in gear or not in gear. The second point is that the proper mechanical arrangement of the pump and the main engine requires careful consideration during the design of the aircraft. Here coordination of effort must be made. The rocket system should be considered as integral to the whole propulsive system, rather than as auxiliary equipment attached to the aircraft after the aircraft has been designed.

Included in the same category of main engine drive are, perhaps, the separate auxiliary engine drive and the gas-turbine drive using combustion gas from the combustion chambers of the turbojet or the main gas-turbine power plant. The scheme with the auxiliary engine is handicapped by lack of a supercharged engine of small output for altitude operation. However, in general, this type of pump drive is complicated and must be considered as expedient only when the directly connected mechanical drive discussed in the preceding paragraph is not available.

If the rocket is the only power plant on the aircraft, then the pumping power must be furnished by independent means. At present, the most promising drive for this purpose is a small gas turbine directly coupled to the centrifugal pump. The gas turbine is driven by combustion products from either an auxiliary small combustion pot or those bled off from the main rocket combustion chamber. This system is called the turborocket. In either case, the combustion products from propellants are generally too hot for safe, long-duration operation of the small gas turbine and have to be cooled by water injection. If hydrogen peroxide is used, then such injection of coolant is not necessary, as the decomposition products of this propellant are at relatively low temperature. The flow through the pumping turbine is roughly 2-3% of the total

flow. Thus, for small thrust of the order of a few thousand pounds, the axuiliary combustion pot may have metering difficulties, especially when bicomponent propellants are used. In general, these systems have the advantage of independent location of the pumping plant and the main rocket motor. For aircraft or missile applications, this freedom is sometimes very desirable. Furthermore, the system is completely independent of altitude and can be operated even in vacuum.

A similar gas-turbine pump drive has been developed by R. H. Goddard. In this system the gas-turbine wheel is driven directly by the rocket jet. This is achieved by bringing the edge of the wheel into the exhaust jet while letting the greater part of the wheel remain outside the jet. Thus the blades of the turbine are in contact with the high-temperature exhaust only a fraction of the time. Hence proper cooling can be accomplished in spite of the very high exhaust temperature. A preliminary estimate shows that the equivalent propellant consumption due to the presence of a gas turbine in the main jet is about the same as for the turborocket drive. However, here the advantage of independent location of the pumping plant and the main rocket motor is lost.

Another propellant-feed method is to combine the pumping plant and the rocket motor into a single unit, such as the centrojet. Here the rocket motor is in rapid rotation. At present, although the possibility of this system has been demonstrated, its perfection as an efficient, dependable unit has to await more fundamental research on the combustion and the cooling of a rotating combustion chamber. However, its inherent compactness and simplicity demand further investigation in these directions.

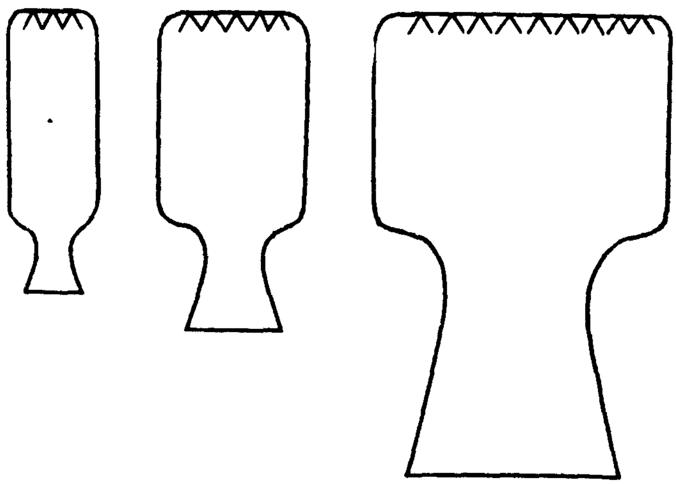
It has also been suggested that a windmill might be used for the pump drive. For high-speed flight, it is more efficient to use a ducted windmill, so that the tip speed of the blades can be kept low. At the design altitude, the equivalent propellant consumption due to the presence of the windmill is approximately four percent of the total propellant flow; thus, it is comparable with the turborocket system. However, the output of the windmill is dependent on the air-flow condition. It seems difficult to have a design that will operate efficiently at a wide range of altitudes. Ultimately, the size of the windmill will be prohibitive at extreme altitudes. Therefore, this type of drive must be considered as a special device with limited applicability.

3. *Motor Construction and Design.*

The two main problems in motor design are the optimum propellant injection for efficient combustion and the cooling of the motor walls for long-duration operation.

The problem of propellant injection is closely connected to the geometrical shape of the combustion chamber. The aim is to utilize fully the volume of the combustion chamber, so that combustion is completed with the smallest possible dimension of the motor. This is important not only for reducing the weight of the motor, but also for reducing the surface of the motor walls and thus decreasing the amount of heat that has to be taken away by cooling. At present, little is known about the detailed flow pattern in the combustion chamber, and it is difficult to ascertain whether the combustion chamber is fully utilized. For instance, if one adopts the concept that the combustion is only a function of the length of time the propellant particles stay

in the chamber, then the length of the chamber should be kept at almost constant value when the thrust rating of the motor is varied, as shown in Fig. 6. However, the path of the gas in the combustion chamber does not necessarily follow the axial direction. In other words, the length and the duration of flow of gas depend upon the flow pattern in the chamber. The simple procedure as shown in Fig. 6 is too crude and does not lead to the most efficient design.



→ Increasing thrust

Figure 6

For long-duration operation, i.e., duration in excess of ten seconds, the motor wall must be cooled to maintain the structural strength. There are three possible methods of cooling the motor: (a) part or all of the propellant passes through a jacket which encloses the motor and the nozzle walls, such as the liquid rocket motor developed by Aerojet Engineering Company in this country; (b) part of the propellant is injected with a very small velocity along the walls, and a liquid film is created over the surface, as a shield against the hot gas, such as in the V-2 motor; (c) part of the propellant is made to seep through a wall of porous material and the wall is cooled by vaporization of the liquid.

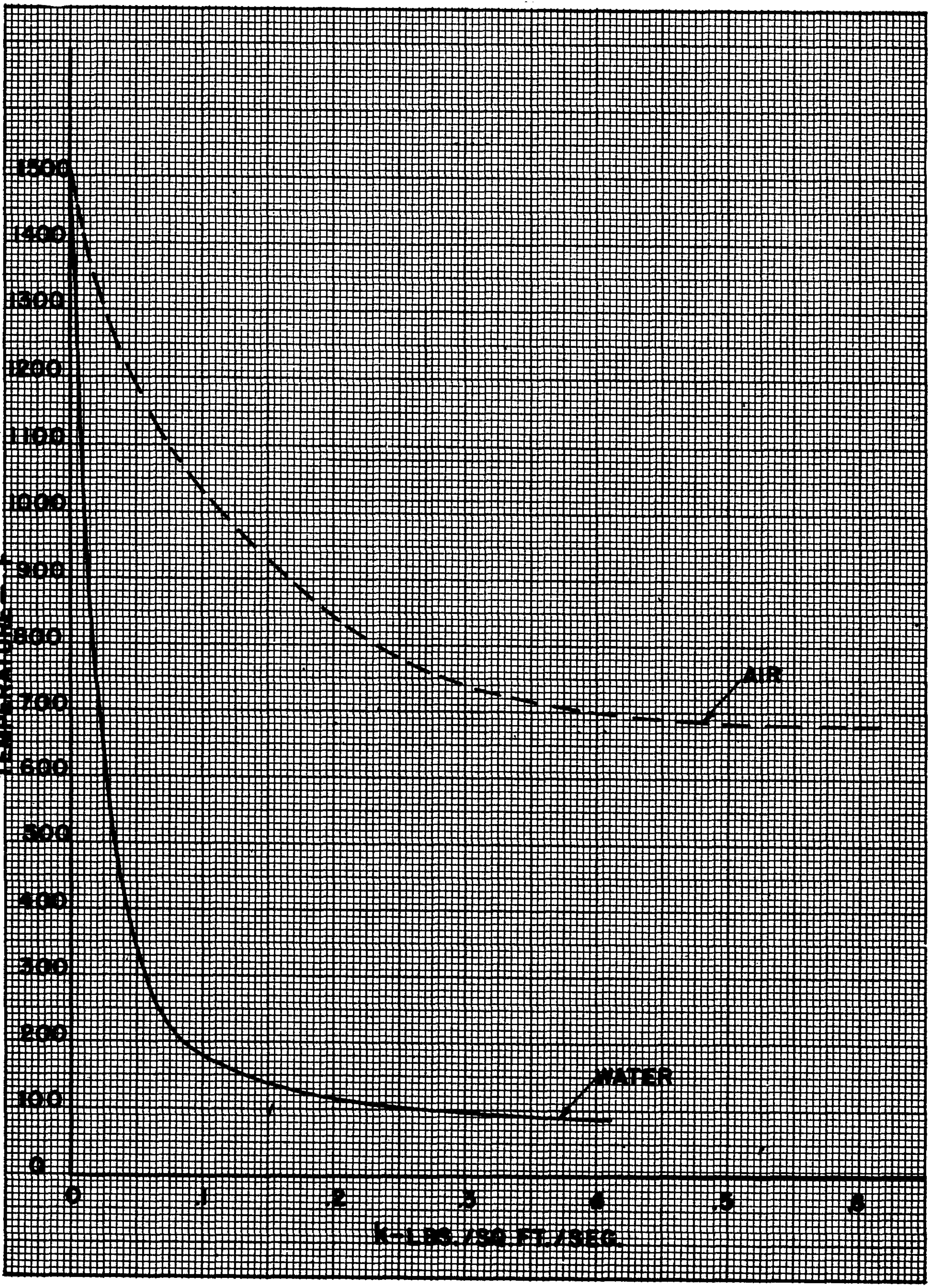
The first method is the so-called regenerative cooling. The difficulties which might be encountered are the local stagnation of the cooling flow and thus hot spots in the wall, and insufficient available quantity of coolant. An example of the latter difficulty is the case of gasoline and liquid-oxygen propellant at optimum mixture ratio, with gasoline as coolant. This difficulty, it seems, would rule out the gasoline and liquid-oxygen combination as a practical propellant. Of course, one possible remedy is to coat the wall with a layer of heat-resisting and heat-insulating ceramic material, so that the rate of heat flow is reduced. The development of such a ceramic coating is, however, only in its beginning. Much research has yet to be done.

The second and the third methods of cooling are similar in principle, in that both depend upon the heat of vaporization to absorb the heat transferred to the wall.

Of course, part of the propellant evaporated will particiapte in the combustion, but the combustion can hardly be as efficient as for the propellant injected through the normal injector into the main combustion zone. Thus a certain loss and an increase in propellant consumption is inevitable. On the other hand, the loss might be quite small and unimportant. By comparing these two methods of evaporative cooling, the more uniform distribution of coolant by the porous walls leads to a more efficient utilization of the liquid. Initial experiments (Fig. 7) indicate that this method is very effective. However, both methods (b) and (c) should be developed for more definite comparison.

4. *Concluding Remarks.*

It is seen that while the future development in solid-propellant rockets leans more heavily on the improvement in the propellant characteristics, the development of liquid-propellant rockets depends more on research on the mechanical design. The design of efficient large-scale rockets for heavy missiles requires especially intensive engineering work on the combustion, cooling, and pump problems.



1500
1400
1300
1200
1100
1000
900
800
700
600
500
400
300
200
100
0
TEMPERATURE — °F
AIR
WATER
0
.1
.2
.3
.4
.5
.6
K—LBS./SQ.FT./SEC.

Figure 7

PART V

HIGH TEMPERATURE MATERIALS FOR AIRCRAFT PROPULSION

BY

Pol Duwez

PART V

HIGH TEMPERATURE MATERIALS FOR AIRCRAFT PROPULSION

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INTRODUCTION

The basic requirements for materials to be used in aircraft propulsion devices depend on the type of engine considered. To discuss these requirements it is convenient to consider separately the three principal types of engine, i. e., turbojets, ramjets and rockets. This subdivision into three classes is justified by the fact that the range of temperatures to which the materials are subjected is different in each type of engine. In a gas-turbine power plant, temperatures from 1200° to 1600°F are at present encountered. In a ramjet unit, a temperature range of from 2500° to 3500°F is generally considered possible. The case of the rocket motor is the most extreme, with temperatures as high as 5500°F in the combustion chamber.

MATERIALS FOR GAS-TURBINE POWER PLANTS

1. *High-Temperature-Resisting Alloys for Gas-Turbine Power Plants.*

The critical part of a gas turbine, as far as materials are concerned, is undoubtedly the blade. Very extensive work has been done during the last few years in order to improve the high-temperature properties of alloys, and meet the severe conditions of stress and temperature present in a modern gas turbine. Among the most important variables to consider in the use of a given material are temperature of operation, maximum stress, allowable deformation during the expected life of the unit, resistance to corrosion and erosion. Other factors, like weldability and machineability are also of importance. A large number of alloys have been produced and tested for applications in which the temperature of the blade itself is between 1200° and 1600°F. The laboratory tests include generally high-temperature strength, time required for rupture under certain conditions of stress and temperature, rate or creep under constant stress and temperature, endurance limit at high temperature, damping capacity and Charpy or Izod impact tests at high temperature. Other physical constants like thermal expansion and termal conductivity have also been measured. As a rule, the most promising alloys on the basis of these laboratory tests were then tested in an actual unit. It is not as yet clearly established which one of the above-mentioned properties is the most

important. Many failures have been encountered in practice that could not be clearly connected to any particular weakness of the materials as shown in laboratory tests. The correlation between actual performance and physical properties is still an important problem.

A general survey of the physical properties of the high-temperature-resisting alloys available at present shows that both tensile strength and creep strength decrease very markedly in the range of temperature between 1200° and 1600°F. The order of magnitude of the maximum safe stress for operation at 1500° to 1600°F is 7000 psi. The best alloys developed so far have, under those conditions, a creep rate of less than 1% in 10,000 hours, which is generally recognized as a safe upper limit. At higher temperatures, very few results have been obtained, and only tensile strength values are available for a few alloys. The order of magnitude of the decrease in strength is from approximately 50,000 psi at 1600°F to about 10,000 psi at 2000°F. From these results it seems logical to conclude that the allowable stress at 2000°F corresponding to an acceptable rate of creep would be much too low to meet the requirements of the present gas-turbine design. This drop in strength with increasing temperature is the obvious consequence of the small margin left between the required service temperature and the melting point of the alloys. More research work on high-temperature-resisting alloys will undoubtedly improve their performance for temperatures below 1600°F. There seems to be little chance, however, that alloys can be developed for higher temperatures.

The evolution of the gas turbine has been marked by continuous increase in inlet gas temperature, which resulted in a corresponding increase in efficiency. If the field of application of alloys is really limited to a certain upper limit in temperature, consideration should be given to other types of material or other methods of design.

The use of ceramic materials with high melting points, would probably raise the temperature limit to 2000°F or higher. Another solution would be to provide an appropriate cooling to the parts of the engine exposed to high temperature. This approach to the problem will eventually require the development of appropriate materials and should be considered from that point of view.

2. *Ceramic Materials for Gas-Turbine Power Plants.*

In the broadest sense of the word, ceramic materials include all materials composed of various combinations of oxides properly mixed together, processed by slip-casting or by pressing, and fired at a sufficiently high temperature to insure a proper bond between particles. This bond is generally obtained by fusion of a small portion of the constituents. The structure of a ceramic material is essentially heterogeneous and consists of crystals imbedded in a glassy matrix, the relative proportion of the two phases varying within wide limits. The essential differences between the structure of a ceramic material and that of a metal are, first, the existence of a vitreous phase in the ceramic material, which is absent in a metal; second, the fact that the crystals in a ceramic material are of the ionic type in which the stability of the lattice is due to electrostatic forces, while the metal crystals are made of neutral atoms linked together by a special kind of bond which is typical of the metallic state. The lack of plasticity in ceramic materials is inherent to their structure, and is likely to remain one of the most

important drawbacks in their use. These fundamental concepts should be borne in mind if successful use of ceramic material is to be made for mechanical devices. Extensive studies should be made in an effort to correlate what is known of the physics of the solid state, and apply it to the study of ceramics. This would give the basic information so badly needed to plan a logical program of research.

To be of any interest to the gas-turbine designer, ceramic materials should have sufficient strength at temperatures above 1600°F. Very little is known of the variation of strength of a ceramic material with increasing temperature. A few tests have indicated that the creep behavior is similar to that of metals. This similitude in the shape of the flow-versus-time curve at a constant temperature does not necessarily mean that the mechanism of flow is the same in the two cases. Creep in metals is mostly due to plastic flow inside the crystals, while it is very probable that a viscous flow takes place in the vitreous phase of a ceramic material. In addition to the strength requirements, ceramic materials for gas turbines should also have a sufficient resistance to variations in temperature. The thermal stresses induced in a material during a transient state of heat transfer depend on the thermal conductivity, termal expansion and modulus of elasticity of the material. These physical constants should be measured accurately and correlated with the structure of the material. At present, very little information exists on this subject. As a matter of fact nothing is known on the variation of modulus of elasticity of ceramic materials with temperature.

The problem of thermal stresses is somewhat complicated by the fact that the structure of the material is heterogeneous. The physical properties of the crystalline and of the vitreous phases being different, local temperature stresses will exist throughout the material, even in a state of thermal equilibrium. Microscopic cracks will eventually develop in the material and will ultimately bring failure. This phenomenon is likely to become a new type of fatigue due to temperature reversals and will require a very detailed study.

An intermediate step in the trend toward using ceramic materials in a gas-turbine power plant is the development of a ceramic coating on metals. Satisfactory coatings have been obtained, and are expected to serve the dual purpose of decreasing the corrosion rate and increasing the resistance to creep. The coating is only a few thousandths of an inch thick and is not intended to serve as a thermal shield.

A further step in combining the properties of metals and ceramics is to produce a composite turbine blade having a metallic center core and an outside ceramic shell. In this case, the existence of a temperature gradient in the ceramic shell would allow an appreciable increase in gas temperature without increasing the temperature of the metalic part of the blade. The ceramic layer would act as a thermal shield, and would not be subjected to an excessive stress. The technique of powder metallurgy has been used for the production of such composite materials. By properly selecting the metal and the ceramic powders, specimens have been obtained, thus showing the feasibility of the method.

Along the same general line some investigators have proposed the addition of ceramic powder (preferably a pure oxide) into a metal powder. The mixed powders are then compacted and sintered at high temperature. It is believed that the creep properties of the mixed material thus obtained are superior to those of the

corresponding pure metal. No actual experimental data is available at present, and further research is necessary before any comment can be made on the chances of success of this method.

The introduction of ceramic materials in a gas-turbine power plant will undoubtedly require some changes in design. These changes should take into account the characteristic features of the ceramic materials. The shape of the blade should be determined with special consideration for the methods of manufacturing. For these materials, the difficulty of obtaining close tolerances is aggravated by the fact that machining is at present very difficult, if not impossible. The method of joining the blade to the motor will be a problem in itself. The maximum stress in service should be reduced as much as possible, and stress concentrations should be minimized. Substituting a ceramic blade for a metal blade in an existing unit would probably be unsuccessful because it would subject the material to unduly severe conditions. In order to assure the maximum chances of success, research on ceramic materials for gas-turbine power plants should be integrated into the more general problem of designing a high-temperature unit in which the advantages of the ceramic would be exploited and its disadvantage minimized.

3. *The Cooling of Gas-Turbine Parts and the Material Requirements.*

The cooling of the parts of a gas turbine exposed to high temperature constitutes a possible answer to the problem of increasing the inlet gas temperature. If cooling is included in the design, the material problem can be considered from a different angle. Essential properties like creep at high temperature become secondary, and other properties like thermal conductivity and thermal expansion become important.

Besides the conventional method of cooling a turbine blade by inside circulation of a liquid, two less orthodox methods have recently been proposed, for which special materials will be needed. In one of these methods the coolant is a gas flowing inside the blade and being ejected along the trailing edge. The blade in this case is a hollow thin-walled tube with an airfoil cross section. The second method of cooling recently suggested consists of making the part to be cooled of a porous material, so that the cooling fluid can be forced through it. In this case the cooling fluid may be either a gas or a liquid. An analysis of the method and a few preliminary experiments have demonstrated the efficiency of this method of cooling. When a liquid is used as a cooling fluid, the temperature of the porous material can be maintained in the neighborhood of the boiling point of the liquid without using an excessive amount of coolant. If this technique is to be applied to turbine blades, the metallurgist is faced with the problem of developing porous alloys having the required physical properties. Since the temperature of the metal will be relatively low, the creep requirements will be much easier to satisfy than in the case of an uncooled blade. The tensile strength of porous metal, however, is relatively low even at room temperature, but if the cooling is as energetic as it is expected, the blade may be kept at a sufficiently low temperature, so that its strength is altogether greater than that of any alloy at temperatures above 1600°F. The endurance limit and the damping capacity are also to be considered. As far as erosion and corrosion are concenred, it is probable that the film of liquid on the surface of the porous metal will have a favorable influence.

Porous metals are obtained by compacting and sintering metal powders. The porosity is obtained by adding to the powder a substance which evaporates or burns during the heat treatment of the compact. Such a technique is already used in the making of porous bearings, but is limited to very few alloys. For gas-turbine applications, the requirements will be much more severe than those encountered in the present applications. This problem opens a very wide field of research. Powder metallurgy is still a new science, and most of the basic phenomena remain without a correct quantitative explanation. Curiously enough, the fundamental problems involved here are essentially the same as those encountered in ceramics, both sciences dealing essentially with chemical reactions in the solid state.

While it is realized that the cooling problem is essentially a design problem, the above considerations indicate it will give rise to extensive research in the field of materials.

MATERIALS FOR RAMJET POWER PLANTS

The ramjet power plant is, without doubt, the simplest type of engine for jet-propelled aircraft. In contrast with the turbojet engine there are no moving parts in a ramjet, and the problem of blade materials is therefore eliminated. There are other material problems however, mostly due to the relatively higher temperature of operation. The gas temperature in the combustion chamber, and in the nozzle of the ramjet engine, may be as high as 3500°F. Fortunately the pressures involved are not very high, and as a consequence, the stresses to which the structure is subjected can be kept relatively low.

At the present time, ordinary stainless steel is used in the construction of combustion chambers for ramjets. This solution appears quite feasible if the temperature of the metal sheets does not exceed some 1200°F. In order to maintain the temperature at this level, a certain amount of cooling of the walls of the chamber is required. This is obtained by a circulation of cold air around the outside of the chamber. The most important drawback in the use of steel is the difficulty in avoiding warping in service. This fact is most troublesome in the case of units intended to operate over long periods of time as compared with those required in aircraft propulsion. The main reason for the uneven deformation of the steel walls is the existence of thermal stresses introduced by a nonuniform temperature distribution. It is probable that parts of the structure which are stressed at high temperatures yield plastically, and upon cooling a permanent localized deformation sets in and causes warping. Therefore, thermal stresses should be reduced to a minimum. For a given design, these stresses depend on the thermal conductivity, and on the thermal expansion of the material. A careful choice of the material may therefore reduce their effect. Unfortunately, thermal conductivity and thermal expansion are of the same order of magnitude for most of the high-temperature-resisting alloys, and it appears that only slight improvements can be expected in the future.

The reduction of the metal wall temperature in a ramjet unit may be attained by using a ceramic coating or a ceramic liner. Ceramic coatings have been developed for protection of the steel exhaust pipes of aircraft engines. These coatings are relatively thin (about ten-thousandths of an inch), and provide a reflective surface for the

infrared rays of the gas. A ceramic liner of low conductivity and sufficient thickness would constitute a more efficient solution, but it would also be a more complicated problem. Ceramic bodies with a sufficiently high melting point to stand the combustion temperature can easily be found. The difficulty is to fabricate a large cylinder with relatively thin walls, having a low density. Besides, such a liner would obviously be mechanically weak, and would fail by cracking when exposed to variations in temperature. To avoid these difficulties, it might be necessary to have a liner made of separate small pieces, each piece being strongly bonded to the metal structure. The question of finding a suitable bond between metal and ceramic is therefore of primary importance.

The presence of a ceramic liner on the walls of a ramjet combustion chamber might have a favorable effect on the combustion of the fuel-air mixture, because it would maintain the walls at a relatively high temperature. This factor has not been experimentally verified yet, and its importance is not clearly established.

MATERIALS FOR ROCKET POWER PLANTS

1. *General.*

The importance of the material problem in the construction of long-duration rocket-motor power plants does not need to be stressed. The large amount of heat generated by combustion at very high temperature in a small volume makes it apparent that no easy solution is to be expected. As in the case of a gas turbine, there exist two different approaches to the material problem. The first is to find materials that would stand the high temperatures developed in the unit, without introducing into the design any special cooling device. The second is to try to decrease the temperature of the materials as much as possible by means of cooling. At the present time the use of the uncooled rocket motor is limited to applications in which the duration of operation does not exceed from 30 to 60 sec. Its interest for aircraft propulsion devices is therefore restricted to assisted-take-off units. The liquid-cooled type of engine is required for aircraft power plants. Although both metals and ceramic materials are very often used simultaneously in a rocket motor, it is convenient to discuss them separately.

2. *Metals and Alloys for Rocket Power Plants.*

In the design of a liquid-cooled combustion chamber, the choice of a metal is primarily determined by its strength at high termperature, its thermal conductivity, and its resistance to corrosion. The strength at high temperature does not have the same importance it has in a gas turbine. In fact, the stress in a jacket-cooled cylindrical chamber can be kept low, and creep at high temperatures is not often a cause of material failure. The thermal conductivity is an essential property, and should be as high as possible in order to maintain the inside wall temperature at a reasonable value. The corrosion resistance is very often the decisive factor in the choice of a metal for a liquid-propellant rocket motor. Most of the propellants used at present are highly corrosive. Corrosion is most critical in the case of the combination nitric acid - anilne, for which only stainless steel, pure aluminum, or chrome-plated alloys can be used. Other propellants, like nitromethane and hydrogen peroxide are unstable when in contact with some metals. In general, however, these factors limit the number of metals

which can be used with each propellant combination, but do not constitute a very critical barrier in the development of rocket motors.

As a rule no serious difficulties are encountered at the present time in building a liquid-cooled rocket combustion chamber, as far as materials are concerned, providing the metal temperature can be kept below a certain maximum value by a correct cooling technique. This last requirement deserves some special consideration. Cooling the combustion chamber consists essentially in removing a certain quantity of heat from the metal walls by raising the temperature of the cooling fluid. In aircraft applications, because of weight consideration, no auxiliary fluid can be carried for the purpose of cooling. One of the propellants must therefore be used for cooling. The maximum amount of heat which can be absorbed depends on the heat capacity of the propellant and the maximum safe temperature to which it can be heated without danger. In some cases, as in a nitric acid - aniline motor, either one of the propellants can absorb enough heat without reaching a dangerous temperature. In other cases, (nitromethane, for example) the temperature of the fluid after its passage through the cooling coil is found to be dangerously high, as a consequence, the design of a regeneratively cooled motor having metal walls becomes a very difficult problem. The logical solution is to line the combustion chamber with a material of low heat conductivity and high melting point, in order to reduce the rate of heat transfer through the walls. Ceramic materials are being considered for this particular application. The characteristic properties of a satisfactory ceramic material for combustion-chamber liners will be discussed in the next section.

The nozzle of a rocket motor is undoubtedly the critical part of the unit as far as materials are concerned. The erosion of the throat by the hot gases flowing at high velocity is still a rather obscure phenomenon. Studies of many nozzle failures due to erosion have shown that in all cases the temperature of the nozzle, at least at the point of failure, must have reached rather high values. It seems that erosion occurs only if the surface in contact with the gas reaches some critical value. Nozzles made of very soft material like aluminum have been used successfully when properly cooled. The hardness of the material at room temperature is not a criterion. A sufficient hardness at the temperature the surface of the nozzle is supposed to reach during operation seems to be required. Here again the choice of material is related to the amount of cooling provided.

For short-duration rocket units using solid propellant as a fuel, a solid copper nozzle properly chrome-plated has been used successfully. Refractory metals have been tested and molybdenum inserts in the throat of a copper nozzle are quite satisfactory.

The method of cooling by injecting a fluid through a porous material seems to offer an interesting solution to the nozzle problem. In this case, the inside of the nozzle is made of porous metal, and a certain quantity of one of the liquid propellants is injected through it. It appears that the quantity of fluid required is small enough to justify the feasibility of the technique. In the case of a solid-propellant unit, a small quantity of water could be carried in the unit, and injected through the throat of the nozzle. The difference in pressure between the chamber and the throat would be used to force the water through the porous metal. The penalty in weight for carrying the water might be counterbalanced by the elimination of the solid copper nozzle.

If successful, this new method of cooling will give rise to a systematic research on the development of porous metals for nozzles of rocket motors.

In planning future research on metallic materials for rocket-motor power plants, careful consideration should be given to the conditions under which these materials are being used. Since cooling appears essential for long-duration aircraft applications, the actual temperature of the metal can be maintained within reasonable limits. There is, therefore, no need for alloys having exceptionally high strength at high temperature. Not too much emphasis should be placed on creep properties at high temperature. On the other hand resistance to corrosion, low thermal expansion, and especially high thermal conductivity, should be the criteria of quality.

3. *Ceramic Materials for Rocket Power Plants.*

The use of ceramic materials in the construction of rocket motors has so far been very limited. Their use would certainly be justified since they combine a high melting point with a low thermal conductivity. A ceramic liner in the combustion chamber of a rocket motor would eliminate or at least simplify the problem of cooling. For rather short-duration applications, it is obvious that an uncooled motor is advantageous from the production point of view. For aircraft rocket power plants, an uncooled motor has less chances of success, but in this case the ceramic liner may be used to reduce the heat transfer to the cooling fluid. This factor is of special interest with some propellants like nitromethane for which, at present, no regeneratively cooled motor can be safely operated because of the danger of overheating the propellant in the cooling coil.

The type of ceramic material required in a rocket motor may depend on a number of factors. In general, however, it can be expected that only very high-melting-point ceramic bodies have a chance of success. A temperature of the order of 3600°F may be taken as a lower limit for the melting point requirement. This consideration eliminates most of the commercially available ceramic materials. The most promising ceramic bodies seem to be those made of pure oxides, or combinations of oxides, with melting points above 3600°F. Among these oxides the most interesting seem to be those of aluminum, beryllium, zirconium, and thorium. The production of such ceramic materials having very high melting points presents great difficulties because of the high firing temperature required. It is not common practice in the ceramic industry to operate furnaces above 3000°F. Test specimens will have to be prepared in laboratory furnaces especially built to provide the necessary firing temperature.

To be used as a liner for a combustion chamber, a ceramic material does not have to retain a very high strength at high temperature since it is completely supported by the outside metal structure. The liner, however, should have a sufficient resistance to rapid variations in temperature. This condition may be more difficult to satisfy than that of having a high melting point. A careful study should be made of the conditions under which a ceramic material fails when subjected to changes in temperature. When the critical factors of this problem are known, the material could be systematically improved.

Ceramic liners have already been used with some success. A zirconium oxide ceramic, for example, can be used in a nitric acid - aniline motor for several minutes

without failure. Much improvement can be expected in the near future, and the development of a satisfactory ceramic chamber liner for long-duration aircraft power plants is not too far remote.

The problem of a ceramic nozzle is by far more difficult. The rather discouraging results obtained so far emphasize the complexity of the question. In general, ceramic nozzles fail by cracking, and show very definite signs of erosion. The very high rate of heat transfer in the nozzle throat is without doubt the reason for the failure. The thermal stresses introduced in the material by an excessive temperature gradient produce cracking. As a consequence of the low thermal conductivity of the ceramic, the temperature of the surface in contact with the gas is very high, and probably above the critical value where erosion cannot be avoided. The important physical properties of the material in this problem are melting point, thermal conductivity, thermal expansion, and some less accurately defined characteristics connected with erosion at high temperature. The relative importance of these variables should be carefully studied in connection with what is known about the heat transfer in the nozzle of a rocket motor. This basic information will then serve as a guide in the search for new materials for nozzle construction.

MATERIALS FOR ATOMIC ENERGY AIRCRAFT POWER PLANTS

Without knowing exactly what will be the design of the future atomic power plant for aircraft, it is difficult if not impossible to speculate on the problems expected in the field of materials. It is very probable, however, that temperatures at least as high as those developed in the present rocket motor will be encountered. In the case in which the atomic energy is used in connection with a thermodynamic working fluid like hydrogen, the temperature in the chamber of the rocket motor is expected to be of the order of 4000°F. It is not known yet how the heat generated by the distinegration will be transferred to the working fluid. It can be expected, however, that a very large release of energy within a small volume will produce extremely high temperatures. It seems very improbable that any material will be found to withstand such temperatures, and the solution may be achieved only by some methods of cooling. For aircraft propulsion applications, for which the weight consideration is of great importance, the amount of coolant should be reduced to a minimum. The design of highly efficient heat exchangers will undoubtedly call for special materials.

Another type of problem which appears to be of primary importance is concerned with materials for protection against radiation. If atomic energy is used in an aircraft power plant, an adequate shield will have to be provided for the pilot. Highly absorbent materials are essentially heavy. In order to minimize the dead weight carried for radiation protection, materials should be developed in which a compromise is made between the two contradictory requirements of achieving a high absorption for radiation and a relatively low density.

In the study of materials for atomic energy development, it will be essential to consider the nuclear properties of the materials. The engineer will certainly have difficulties in carrying on useful research without the help of the physicist. This rather unorthodox approach to metallurgical problems opens an entirely new field of research. There is a strong possibility that progress will be slow because of the lack of scientific personnel having a sufficient background in nuclear physics combined with an engineering knowledge of materials.

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CONCLUSIONS

The problem of high-temperature-resisting materials is one of the most essential in the future improvements in aircraft power plants. In the past, most of the effort has been devoted to the development of materials for gas-turbine blades. Quite satisfactory alloys are now available to withstand temperatures up to 1600°F. As a consequence of the relation between the efficiency of a gas turbine and its operation temperature, it is believed that the evolution of the gas turbine is toward higher and higher temperatures. The physical properties of the present alloys at temperatures above 1600°F are too low to expect any appreciable increase in operation temperature. The two apparent solutions to the problem are the use of ceramic materials, or the cooling of the blade. These two solutions give rise to new problems in the field of materials.

Concerning special materials for ramjet and rocket motors, very little systematic research has been done. In studying high-temperature-resisting materials, consideration should be given to the special conditions encountered in these types of motors. The approach to the problem is essentially different from that of the gas-turbine blade. Ceramic materials and cooling should also play an important part in this program of research.

To make any research successful, a sufficient knowledge of the basic principles involved in the problem is absolutely necessary. This is particularly true in the field of materials. The physics of the solid state should be the foundation on which future research should be based. While the metallurgist has already made efficient use of the present knowledge of the structure of matter, its importance should be emphasized, particularly in view of the increasing interest in relatively less explored fields like those of powder metallurgy and ceramics. The introduction of the use of atomic energy in engineering developments brings the final argument for the need of basic research in the field of materials.

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An analysts of the design, performance and thermodynamic cycles of the gas turbine propulsion units includes a review of reciprocating engines, compound engines, free-ptston engines, gas turbines, and turbojet engines. The experimental and theoretical performance of pulsejet engines have been considred and possible improvements are suggested. The performance of ramjets and their design problems are summarized. Future trends in the design and development of solid and liquid fuel rocket engines are indicated, and a study of high temperature materials required for jet and rocket engines ts included.

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